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OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME AUXILIARY COMPUTING FACILITY APOLLO 9 FLIGHT ANNEX

FLIGHT ANALYSIS BRANCH

MISSION PLANNING AND ANALYSIS DIVISION

MANNED SPACECRAFT CENTER

HOUSTON, TEXAS

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PROJECT APOLLO

OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME AUXILIARY COMPUTING FACILITY APOLLO 9 FLIGHT ANNEX

By H. L. Sanders
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February 17, 1969

MISSION PLANNING AND ANALYSIS DIVISION
NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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HOUSTON, TEXAS

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DEFINITIONS

Blackout Data	Blackout data include the ground elapsed time, latitude, and longitude of each entrance and exit of the blackout region for both S-band and VHF.
Entry Data	Entry data are generally intended to imply inertial velocity, flight-path angle, latitude, and longitude at the 400, 000-foot altitude, blackout data, maximum g-load, landing point, and footprint data.
Entry Interface	Entry interface is considered to be the dividing line between the tenable atmosphere and space and is considered to occur at an altitude of 400, 000 feet.
Footprint Data	Footprint data include the latitude and longitude for both a zero and a full lift entry.
Horizon Monitor	Horizon monitor refers to a spacecraft attitude which is maintained 31. 7 degrees above or below the visual horizon.
Maneuver Data	Maneuver data consist of those quantities which are necessary to completely define a maneuver. These quantities may include engine ignition, imparted velocity, guidance mode, etc.
Radar Data	Radar data include the ground elapsed time of acquisition and loss, minimum and maximum elevation angles, minimum range, tracking duration, and revolution number of each pass over a station.
RTCC State Vector	The term "RTCC State Vector" is meant to include the following quantities transmitted to the RTACF from the RTCC. 1. Vector identification 2. Lift-off time in GMT 3. Vector time in GMT 4. Position vector components 5. Velocity vector components 6. Revolution number 7. Spacecraft weight
REFSMMAT	REFSMMAT relates the Besselian coordinate system to the IMU stable member coordinate system.

NOMENCLATURE

ABDP	Apollo Block Data Program
ACR	Auxiliary Computing Room
ACRA	Atlantic continuous recovery area
AGOP	Apollo Generalized Optics Program
AGS	abort guidance system
AOS	acquisition of signal
AOT	alignment optical telescope
APS	ascent propulsion system
ARMACR	RTACF Apollo Reference Mission Program
ARRS	Apollo Real-Time Rendezvous Support Program
ARS	Apollo Reentry Simulation Program
CDH	constant differential height
CLA	contingency landing area
CM	command module
CMC	command module computer
COAS	crew optical alignment sight
CSM	command and service module
CSI	concentric sequence initiation
DAP	digital autopilot
DMT	detailed maneuver table
DPS	descent propulsion system
ECS	environmental control system
EMS	entry monitoring system
EMU	extravehicular mobility unit
EPS	electrical power system
ETR	Eastern Test Range

EVA	extravehicular activity
FAB	Flight Analysis Branch
FDAI	flight director attitude indicator
FDO	flight dynamics officer
FTP	fixed throttle point
GEMMV	General Electric Missile and Satellite Multi-Vehicle Program
g.e.t.	ground elapsed time
GMT	Greenwich mean time
G&N	guidance and navigation
GPMP	general purpose maneuver processor
IMU	inertial measurement unit
ITT	International Telephone and Telegraph
IU	instrument unit
IVT	intravehicular transfer
LAB	Lunar Analysis Branch
LCG	liquid cooled garment
LEC	Lockheed Electronics Company
LET	launch escape tower
LGC	lunar module guidance computer
LHe	liquid helium
LiOH	lithium hydroxide
LM	lunar module
LOS	loss of signal
LOST	lunar optical support table

LOX	liquid oxygen
LPD	landing point designator
LV/LH	local vertical/local horizontal
MPSO	Mission Planning Support Office
MPT	mission plan table
MRS	Mass Properties, Reaction Control System, Service Propulsion System Program
MSFN	Manned Space Flight Network
MTVC	manual thrust vector control
OMAB	Orbital Mission Analysis Branch
OST	optical support table
PAO	public affairs office
PGNCS	primary guidance and navigation control system
PLA	primary landing area
PLSS	portable life support system
PTC	passive thermal control
PVT	pressure-volume-temperature
RCS	reaction control system
REFSMMAT	reference to stable member matrix
REM	roentgen equivalent man
RTACF	Real-Time Auxiliary Computing Facility
RTCC	Real-Time Computing Complex
S-IC	first stage of the Saturn V launch vehicle
S-II	second stage of the Saturn V launch vehicle
S-IVB	third stage of the Saturn V launch vehicle
SCS	stabilization control system
SHE	Supercritical Helium Program

SLA	spacecraft lunar module adapter
SM	service module
SPAN	Solar Particle Alert Network
SPS	service propulsion system
SST	star sighting table
T&D	transposition and docking
TLI	translunar insertion
TPI	terminal phase initiation
TPI_o	terminal phase initiation near perigee
TRW	TRW Systems Group

OPERATIONAL SUPPORT PLAN FOR THE REAL-TIME AUXILIARY COMPUTING FACILITY APOLLO 9 FLIGHT ANNEX

By H. L. Sanders
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MISSION OPERATIONS SECTION

TRW Systems Group

1. INTRODUCTION

1.1 Purpose

The Apollo 9 Flight Annex summarizes those aspects of the operational support plan peculiar to the Apollo 9 mission. Included in this document are a description of the support requirements of the Real-Time Auxiliary Computing Facility (RTACF) for the Apollo 9 mission and a brief description of the programs that will be used for this support. It is intended to provide a central source of RTACF mission support information pertaining to this mission.

1.2 Method of Presentation

The Apollo 9 Flight Annex is divided into six sections, the first of which is the introduction. The second section states the main objectives of the mission and briefly outlines the major nominal mission events. The third section contains a description of the RTACF computing requirements for mission support, while the fourth section presents a description of the processors that will be used to satisfy these requirements. The fifth section presents the RTACF nominal mission timeline, and the sixth section lists the personnel assigned to the RTACF in support of the Apollo 9 mission.

2. GENERAL MISSION DESCRIPTION

2.1 Introduction

A brief description of the Apollo 9 mission profile is presented in this section. The nominal mission description and certain tables of discrete events have been extracted in part from the Apollo 9 Spacecraft Operational Trajectory (Reference 1).

2.2 Nominal Mission

Apollo 9, the second manned Saturn V mission, is planned as a 10-day, earth orbital, combined command service module (CSM) and lunar module (LM) operations mission that will demonstrate the capability of the CSM and LM to perform selected functions of the lunar landing mission. It is the first flight test program that includes both manned CSM and LM operations. This mission also features the only extra-vehicular activity (EVA) of the test program prior to the lunar landing mission. Table I contains a listing of the nominal mission events.

The nominal trajectory is based on a 1600 Greenwich mean time (GMT) lift-off from pad 39A on February 28, 1969. The launch azimuth is 72 degrees from true North. The mission profile has been divided into six periods of activity.

2.2.1 First period of activity. - The first period of activity extends from lift-off to 19 hours ground elapsed time (g. e. t.) and consists of the following events:

a. Launch and insertion into earth parking orbit: The Apollo 9 launch vehicle is the Saturn V. The launch phase consists of the complete burns of the S-IC first stage and S-II second stage and a partial burn of the S-IVB. The spacecraft is inserted into a 103 nautical mile, near-circular parking orbit. The sequence of events during launch is given in Table II.

b. Postinsertion activities: Following insertion, the S-IVB maintains the cutoff attitude for 20 seconds before maneuvering the complete configuration to a local horizontal attitude. The S-IVB then imparts an orbital rate to the CSM/LM/S-IVB combination. The flight crew then conducts a series of systems checks and configures the CSM for orbital operations. A pretranslunar insertion (TLI) demonstration is conducted within the first orbit, and the CSM platform is then realigned to the launch orientation in the first period of darkness.

c. Transposition and docking (T&D): At approximately 2:34 (hr:min) g.e.t., the S-IVB assumes a T&D attitude with the CSM pitched 10 degrees below the local horizontal and yawed 15 degrees south from the orbital plane. This T&D attitude was selected for nominal launches in the February 20, 1969 to May 20, 1969 time frame.

At sunrise, approximately 2:43 (hr:min) g.e.t., the CSM separates from the S-IVB/LM with a 1 fps ΔV. The spacecraft launch vehicle adapter (SLA) panels are jettisoned at the time of CSM separation. After coasting approximately 1 minute to a range of 50 feet, the CSM reduces the separation velocity to 0.5 fps, pitches 180 degrees to look at the S-IVB/LM, and nulls the separation velocity. The CSM then closes on the S-IVB/LM.

After reaching the docking interface, the CSM maintains its position relative to the S-IVB/LM for a period of approximately 15 minutes. Having rolled -60 degrees, the CSM soft docks with the LM at approximately 3:00 (hr:min) g.e.t. The LM is repressurized, and hard docking is completed before entry into darkness.

d. LM ejection and separation: The CSM remains docked through the third period of darkness, and at approximately 4:09 (hr:min) g.e.t., the LM is ejected from the S-IVB. The CSM/LM coasts approximately 2.5 minutes before a 3-second service module reaction control system (SM RCS) burn provides permanent separation of the CSM/LM and S-IVB.

e. S-IVB restarts: The S-IVB returns to a local horizontal attitude at approximately 4:35 (hr:min) g.e.t., and preparations begin for the first of two S-IVB restarts. The first restart occurs at approximately 4:44:42 (hr:min:sec) g.e.t. (90 percent thrust). The second restart occurs at approximately 6:06 (hr:min) g.e.t. (90 percent thrust), and the scheduled liquid oxygen (LOX) and liquid helium (LH_e) propellant dumps are completed by 6:27 (hr:min) g.e.t. Reference 2 contains detailed information on the S-IVB activities for this mission.

f. SPS-1: Meanwhile, the CSM inertial measurement unit (IMU) is realigned to the preferred alignment, and at 6:01:40 (hr:min:sec) g.e.t., as the CSM/LM passes over Hawaii, the first docked service propulsion system (SPS) burn occurs. The 5-second, 36.8-fps burn is under guidance and navigation (G&N) control and uses external ΔV targets. Resulting apogee and perigee altitudes are 131 and 113 nautical miles, respectively. No ullage is required. A partial demonstration of the capability of the G&N system to control the spacecraft in a docked configuration during a ΔV maneuver is accomplished during the burn (CSM autopilot stability margin).

2.2.2 Second period of activity. - The second period of activity extends from 19 to 49 hours g.e.t. and includes the second, third, and fourth SPS burns.

a. SPS-2: At 22:12:00 (hr:min:sec) g.e.t., as the CSM/LM passes over the Bermuda tracking station, the first of this period's three long-duration, docked SPS burns takes place. The G&N-controlled, external-ΔV burn is targeted primarily to shift the node eastward but also has a small in-plane ΔV component to raise apogee to 191 nautical miles. The nodal shift, resulting from SPS-2 (and for SPS-3, SPS-4, and the first descent propulsion system burn (DPS-1)), is necessary to open the launch window to approximately 3 hours. For various late lift-offs within the launch window, some of the maneuvers will be reoriented to shift the line of nodes westward. The apogee adjustment of this burn also permits large crew monitoring limits and prevents trajectory problems resulting from a low perigee if large attitude dispersions occur during the burn. Additional trajectory objectives of this burn are to provide continuous SM RCS deorbit capability and to reduce the SM RCS propellant requirements for LM rescue during the rendezvous period by reducing the CSM mass. Approximately 7360 pounds of SPS propellant are consumed during the 110.4-second, 849.4-fps burn. If rigid body transients were not observed on SPS-1, the SPS pitch gimbal will be intentionally misaligned 0.5 degree. The burn partially demonstrates the CSM digital autopilot (DAP) attitude control capability, and the 40 percent amplitude stroking test is performed. No ullage is required.

b. SPS-3: Approximately two revolutions after SPS-2, the CSM/LM passes over Merritt Island tracking station at 25:18:30 (hr:min:sec) g.e.t., and a 2548.1-fps, G&N-controlled, external-ΔV burn is performed which shifts the line of nodes approximately 10 degrees east. Although targeted primarily out of plane to give a large nodal shift, a small in-plane ΔV component raises apogee from 191 to 271 nautical miles and allows for possible large attitude excursions and large crew monitoring limits.

Approximately 18,680 pounds of SPS propellant are consumed during the burn, further increasing the SM RCS capability for use in LM rescue or backup SM RCS deorbit. Not less than 10 seconds after ignition and no later than 30 seconds before termination, a full amplitude stroking test is employed. This test completes the CSM DAP stability margin objective.

c. SPS-4: The final burn of the second period of activity takes place at 28:28:00 (hr:min:sec) g.e.t. over Texas. This burn is also G&N-controlled and nominally targeted to shift the line of nodes 1 degree east. The 28.0-second, G&N-controlled, external-ΔV burn provides 299.8 fps out of plane. Apogee and perigee altitudes (271 and 115 nautical miles, respectively) remain unchanged. If, however, launch is delayed more than 15 minutes, the burn will be retargeted with an in-plane component to adjust apogee (a phasing prior to rendezvous initiation) to return the trajectory to nominal lighting and tracking during the rendezvous.

2.2.3 Third period of activity. - The third period of activity extends from 49 to 60 hours g.e.t. and includes the following events:

a. LM power-up and systems check: The third period, beginning at approximately 49:00 (hr:min) g.e.t., is devoted primarily to LM systems checkout and evaluation. At approximately 42:10 (hr:min) g.e.t., the two LM crew members transfer intravehicularly to the LM. The intravehicular transfer (IVT) will demonstrate the crew's ability to transfer to the LM by this method, and it will verify procedures and premission planning times associated with the transfer. After transfer, the LM crew will power up the LM and conduct a systems check and evaluation. This activity terminates at the beginning of the docked descent propulsion system burn.

b. Docked DPS burn: The DPS ignites at 49:43 (hr:min) g.e.t. and burns for 364.0 seconds (280 seconds at fixed throttle point (FTP)) and is under primary guidance and navigation control system (PGNCS) attitude control. The burn time is sufficient to verify LM PGNCS DAP attitude control capability during a docked thrusting maneuver. The flight crew will also demonstrate manual throttling and termination of the descent engine.

After the DPS has burned for 280 seconds at FTP, the flight crew will manually throttle the DPS during the remainder of the burn and will manually terminate the burn. Crew takeover corresponds to a point at which 124-fps ΔV remain to be gained. Although the burn is targeted to 368 seconds, the burn is terminated 4 seconds early to ensure a manual cutoff.

The LM thrust profile results in 1698.3 fps being used to shift the line of nodes east by 6.7 degrees for a nominal launch time. For late lift-offs, the burn will be targeted to always give an apogee of 270 nautical miles and a perigee of 115 nautical miles. A 10-second LM RCS two-jet ullage is planned.

After the docked DPS burn, which is covered by Merritt Island and Bermuda, the LM systems are powered down, and the LM crew returns to the CSM through the tunnel.

c. SPS-5: At 54:26:16 (hr:min:sec) g.e.t., the SPS burns for 41.5 seconds over the Eastern Test Range (ETR). The resulting 550.5 fps from the PGNCS-controlled burn circularizes the orbit at 133-nautical mile altitude. The resulting circular orbit is the base orbit from which the LM-active rendezvous will be initiated in the fifth period. A 20-second, two-jet ullage is required. For late lift-offs, SPS-5 will continue to be a circularization maneuver and will always last at least 40 seconds.

2.2.4 Fourth period of activity. - The fourth period of activity extends from 60 to 89 hours g.e.t. and is primarily concerned with extravehicular activity.

a. Extravehicular activity preparation: At approximately 60:00 (hr:min) g.e.t., two crewmen will move through the docking tunnel to the LM and power up and check the LM systems. One LM crew member will don the extravehicular mobility unit (EMU) and, after a thorough checkout, will egress the LM at approximately 71:40 (hr:min) g.e.t.

b. EVA: The EVA crewman will move from the LM to the command module (CM) side hatch using the surface translation aids. The EVA crewman will then ingress the CSM. After a brief rest, the EVA crewman will egress the CSM and move back to the LM. Some of the EVA activities are to: (1) evaluate the CM hatch, (2) retrieve and stow collected samples, (3) evaluate EVA light, and (4) perform EVA photography.

c. Post-EVA: Entering the LM through the forward hatch, the EVA crewman will doff the EVA equipment after repressurization of the LM. The portable life support system (PLSS) is then recharged, and the LM systems are shut down before the LM crew does an intravehicular transfer through the docking tunnel.

2.2.5 Fifth period of activity. - The fifth period of activity extends from 89 to 120 hours g.e.t. and includes the rendezvous and the ascent propulsion system (APS) burn to depletion.

a. Rendezvous profile: The rendezvous profile begins with a 5-fps ΔV radially down, small "football" separation by the SM RCS (called a "football" because of the appearance of the LM's motion relative to the CSM; see Figure 1). The small football resulting from the 5-fps SM RCS burn is included to prevent the LM or CSM from wasting attitude propellant while station keeping during platform alignments. In addition, the maximum separation of 2.8 nautical miles is sufficient to prevent recontact of the CSM and LM during the large football phase. One-half orbit after the small football, the LM burns radially up into a larger football from which it then maneuvers into a concentric orbit above and behind the CSM. After a period of coelliptic coast, the LM proceeds through one concentric sequence initiation - constant differential height (CSI-CDH) sequence which sets up the conditions for the nominal LM-active terminal phase initiation (TPI) from below and behind.

Since this is the first manned LM separation from CSM that is of any significant distance, the larger football resulting from the radial burn is included mainly as a safety measure. The football will permit a less difficult LM return to the CSM if the situation dictates. The large LM radial burn, termed the phasing burn, is the only abort guidance system (AGS) controlled maneuver on the nominal mission. For all other LM burns, the PGNCS will be primary, and the AGS will be in a monitor mode.

During the rendezvous, the CSM will orient for and prepare to execute any maneuver that the LM is unable to perform. If this situation arises, the CSM would essentially execute an equal but oppositely-directed maneuver commonly referred to as a "mirror image" maneuver.

b. SM RCS initiated football phase: The fifth period of activity begins with the intravehicular transfer of the two LM crew members at approximately 89:00 (hr:min) g.e.t. After LM power-up and systems checkout, both the LM and CSM IMU's are aligned. The LM alignment is that of a "nominal" alignment for the TPI, and the time on which the alignment is based corresponds to the time of ignition at TPI. The +X CSM stable member is colinear with the +Z LM stable member, the +Y CSM is colinear with the +Y LM stable member, and the +Z CSM stable member is colinear with the -X LM stable member. Following IMU alignments, the CSM and LM undock approximately 25 minutes before the SM RCS separation burn and station keep until approximately 93:05:45 (hr:min:sec) g.e.t. The SM RCS then provides a 5-fps radially down burn to place the CSM into an equiperiod orbit (small football). The maneuver is performed with the -Z body RCS thrusters.

Because of nonnominal burn executions and differential atmospheric drag, the 5-fps ΔV is large enough to prevent recontact of the CSM and yet small enough to permit a simplified abort of the nominal rendezvous. After an IMU fine-alignment rendezvous radar checkout, the LM maneuvers for the AGS-controlled DPS phasing maneuver.

c. LM DPS phasing maneuver: At 93:50:04 (hr:min:sec) g.e.t. after a 7-second, two-jet RCS ullage, the DPS burns at 10 percent thrust for 15 seconds and 40 percent thrust for 10.2 seconds and provides 85-fps ΔV directed radially up. With this phasing burn over the Mercury tracking ship, the LM is placed into an equiperiod trajectory approximately 11.2 nautical miles above and behind that of the CSM and with a maximum LM-to-CSM range (at the horizontal crossing) of 47.9 nautical miles.

The resulting football is planned so that favorable lighting and Manned Space Flight Network (MSFN) coverage, as well as acceptable ΔV's, exists if a 130 degree terminal phase transfer near perigee is required. Such a transfer (TPI_O) would be applicable if the nominal rendezvous profile were aborted prior to the coelliptic maneuver called the insertion burn. Once the decision is made to continue the rendezvous, the LM coasts approximately 111.7 minutes after the phasing burn through the TPI_O position (near perigee of the football) to a position near apogee.

d. LM DPS insertion maneuver: At approximately 95:41:48 (hr:min:sec) g.e.t. following a 7-second, two-jet RCS ullage, the DPS ignites over Guaymas and burns for 24.8 seconds at 10 percent thrust to provide 39.9-fps ΔV. The PGNCS-controlled, external-ΔV burn establishes a near-circular orbit approximately 11.2 nautical miles above that of the CSM. Following the burn, the LM coasts for approximately 40.2 minutes before staging and the CSI maneuver.

e. LM RCS-interconnect CSI maneuver: At approximately 96:22:00 (hr:min:sec) g.e.t., the CSI (a horizontal, retrograde burn) maneuver over Tananarive is scheduled. Staging occurs at the beginning of the 30.6-second, four-jet interconnect, PGNCS-controlled burn. The 37.8-fps ΔV from the maneuver creates a perigee approximately 10.1 nautical miles below that of the CSM.

f. LM APS CDH maneuver: After a 44.5-minute coast, the LM reaches perigee. Following a 4-second, four-jet LM RCS ullage, the ascent propulsion system burns for 3.1 seconds and provides 37.9-fps ΔV nearly horizontal and retrograde. The PGNCS-controlled burn over the Redstone tracking ship establishes a near-circular orbit approximately 10.1 nautical miles below the CSM. At burn termination, the LM trails the CSM by 75.8 nautical miles.

g. LM RCS TPI maneuver: The LM coasts approximately 53 minutes after CDH before reaching the TPI point. Initiated over Tananarive on an elevation angle of 27.5 degrees at approximately 8 minutes into darkness, the TPI burn of 17.6 seconds (four-jet RCS interconnect) and 21.9-fps ΔV is applied primarily along the line of sight. The 130 degree terminal phase requires about 45 minutes (including braking).

Station keeping begins at approximately 98:36 (hr:min) g.e.t. over Hawaii, and docking is assumed to occur at 98:53 (hr:min) g.e.t.

h. APS long-duration burn: After docking, the crew configures the ascent stage for the long APS burn and then returns to the CSM. The CSM jettisons the ascent stage and applies 3.0 fps out of plane to provide a safe separation at APS ignition. At approximately 100:26 (hr:min) g.e.t., the APS thrusts for an estimated 360.0 seconds to propellant depletion. The LM IMU remains unchanged from the alignment for the rendezvous; consequently, the thrust is targeted to a maximum of 45 degrees out of plane to prevent gimbal lock. The resulting 5246.7 fps from the PGNCS-controlled, external-ΔV burn over Guaymas, Texas, and Merritt Island raises apogee to 3237-nautical mile altitude. The burn is preceded by a four-jet, 3-second ullage, and the interconnect remains open through the burn to provide attitude control.

2.2.6 Sixth period of activity. - If the SPS becomes unable to deorbit the CM, the SM RCS will be required to effect the deorbit. The ΔV available from the SM RCS is adequate for this purpose through the first five periods of activity. To reduce the RCS ΔV requirements for deorbit in the sixth period of activity, or, equivalently, to reduce the SM RCS propellant reserved for the backup deorbit, a perigee adjustment is scheduled for the day following rendezvous.

a. SPS-6: At approximately 121:59 (hr:min) g.e.t., the SPS thrusts over Carnarvon for 2.4 seconds under control of the G&N using external-ΔV targets. Preceded by a 20-second, two-jet ullage, the burn yields 62.7 fps and lowers perigee to 95 nautical miles.

b. SPS-7: Two days following the SPS-6 perigee adjustment, the final trajectory adjustment before deorbit occurs (169:47 (hr:min) g.e.t.). The burn, SPS-7, is designed to raise apogee to 210 nautical miles and to position apogee in the Southern Hemisphere. This positioning is desirable because: (1) it allows more free-fall time to 400, 000 feet (assumed entry interface) after the nominal SPS deorbit, and (2) it slightly reduces the RCS-ΔV requirements for backup SM RCS deorbit into the nominal recovery area.

The maneuver scheduled over Merritt Island requires 6.2 seconds of burn and 155.7 fps of ΔV. A 20-second, two-jet ullage is necessary prior to SPS ignition. The burn is G&N-controlled and uses external-ΔV targets.

c. SPS-8: The deorbit maneuver, SPS-8, is scheduled at 238:10 (hr:min) g.e.t., which is slightly less than 3 days after SPS-7. The retrofire occurs approximately 700 nautical miles southeast of Hawaii and is covered by both Hawaii and Redstone. The deorbit burn of 12.3 seconds and 313.4 fps is a G&N-controlled and external-ΔV-targeted maneuver which requires a 20-second two-jet ullage.

d. Preentry sequence: At 9:22:10:12.28 (day:hr:min:sec) g.e.t., the SPS engines will be cut off, and a 15.3-minute coast to atmospheric entry will be started. Approximately 90 seconds following the deorbit burn, the SM will be jettisoned. The crew will manually yaw the CSM 45 degrees out of plane to the north before separation to ensure against possible CM/SM recontact. After CM/SM separation, a short time is allowed for an adequate separation distance, and the spacecraft is maneuvered to the entry trim attitude.

e. Entry: Entry at 400, 000 feet altitude will occur at 33.36 degrees North geodetic latitude and 89.70 degrees West longitude approximately 15.3 minutes after SPS retrofire. Landing of the CM will occur at 9:22:40:48 (day:hr:min:sec) g.e.t. at 30.64 degrees North geodetic latitude and 58.97 degrees West longitude.

Table I. Nominal Mission Events

Mission Event	Time from Lift-off (day:hr:min:sec)	Geodetic Latitude (deg:min:sec)	Longitude (deg:min:sec)	Altitude (n mi)
S-IVB/CSM insertion	00:00:11:15.7	32:36:18.9N	55:52:35.5W	103
S-IVB/CSM separation	00:02:43:00.0	07:31:25.9S	169:05:31.5E	105
LM ejection	00:04:08:57.0	12:20:35.0S	139:03:30.6E	105
First SPS ignition	00:06:01:40.0	31:46:15.0N	155:40:23.9W	110
Second SPS ignition	00:22:12:00.0	27:40:33.4N	63:37:07.9W	110
Third SPS ignition	01:01:18:30.0	33:28:52.4N	79:27:53.2W	113
Fourth SPS ignition	01:04:28:00.0	32:33:31.9N	94:41:16.7W	126
LM systems evaluation	01:16:00:00.0	26:22:57.2S	65:16:09.2W	236
Docked DPS ignition	02:01:42:00.0	33:06:52.5N	84:51:56.4W	113
Fifth SPS ignition	02:06:25:16.0	30:10:33.5N	112:45:53.0W	129
Extravehicular activity	02:23:40:00.0	13:45:32.5S	156:16:04.9W	128
CSM/LM separation (SM RCS)	03:21:05:45.0	24:35:16.6N	32:13:24.6E	129
Phasing ignition (DPS)	03:21:50:03.6	24:55:07.6S	159:59:11.3W	129
Insertion ignition (DPS)	03:23:41:48.1	20:59:48.6N	106:47:45.4W	129
CSI ignition (LM RCS)	04:00:22:00.0	12:34:07.6S	46:47:56.3E	127
CDH ignition (APS)	04:01:06:22.6	12:15:15.7N	145:05:30.2W	128
TPI ignition (LM RCS)	04:02:00:15.0	28:22:15.4S	57:45:58.9E	128

Table I. Nominal Mission Events (Continued)

Mission Event	Time From Lift-off (day:hr:min:sec)	Geodetic Latitude (deg:min:sec)	Longitude (deg:min:sec)	Altitude (n mi)
TPF ignition (LM RCS)	04:02:31:41.7	04:12:00.0N	179:56:00.0W	128
APS/DPS staging	04:03:56:00.0	06:16:08.0S	142:16:38.0E	128
APS burn to depletion ignition	04:04:26:00.0	30:53:52.6N	102:36:53.3W	130
Sixth SPS ignition	05:02:01:00.0	26:43:46.3S	122:18:46.2E	128
Seventh SPS ignition	07:01:49:00.0	28:53:16.1N	78:56:10.8W	94
Eighth SPS ignition	*	*	*	*
Entry interface	*	*	*	*
Drogue parachute deployment	*	*	*	*
Main parachute deployment	*	*	*	*
Landing	*	*	*	*

*Entry quantities not available at time of publication.

Table II. Apollo 9 Launch Vehicle Operational Trajectory Sequence of Events

Event	Time from lift-off, (min:sec)
Lift-off	0:00.0
Maximum dynamic pressure	1:17.250
S-IC inboard engine cutoff	2:19.316
Tilt arrest	2:20.000
S-IC thrust termination and S-IC/S-II separation	2:30.623
S-II stage at 90 percent thrust	2:35.061
Mixture ratio shift to 5.5	2:37.161
Jettison S-IC/S-II forward interstage	3:00.861
Jettison launch escape system	3:05.861
Mixture ratio shift to 4.7	7:02.210
S-II thrust termination and S-II/S-IVB separation	8:44.192
S-IVB stage at 90 percent thrust	8:49.192
Jettison S-IVB ullage cases	8:56.641
S-IVB stage thrust termination - parking orbit insertion - postinsertion ullage	11:35.280

3. APPROVED RTACF SUPPORT REQUIREMENTS

3.1 Introduction

The requirements that the Real-Time Auxiliary Computing Facility will support for the Apollo 9 mission are given in References 3 through 5 and are briefly summarized in this section. These requirements have been assigned to the RTACF as outlined in Section 6 of the Operational Support Plan (Reference 6) and have been discussed and mutually agreed upon by the Mission Planning and Analysis Division and the Flight Control Division. Details concerning the inputs from the Real-Time Computing Complex (RTCC), flight controllers, and required outputs from the RTACF processors to satisfy these requirements are given in Section 4 of this document. These are the requirements that were identified prior to the beginning of flight control simulations; others can and probably will be added during flight control simulations.

3.2 Mode I Launch Abort Requirements

Determine the spacecraft landing points should an abort be necessary during the first 90 seconds of powered flight based on wind profiles from 24 to 1 hour prior to launch.

3.3 General Orbit Phase Requirements

3.3.1 Orbital lifetime. - Determine lifetime of the vehicle (CSM, S-IVB, or LM) in days, hours, and minutes, g.e.t., given a state vector, vehicle weight, and vehicle aerodynamics.

3.3.2 K-factor. - Determine the atmospheric density K-factor, given the vehicle weight, drag coefficient, effective aerodynamics cross-sectional area, and two or more state vectors.

3.3.3 Flight dynamics officer (FDO) orbit digitals. - Compute the FDO orbit digitals, given the vehicle weight, drag coefficient, effective aerodynamic cross-sectional area, and a state vector.

3.3.4 Relative motion. - Determine the post separation, relative motion of the S-IVB, SM, or LM with respect to the CSM or CM.

3.3.5 CSM horizon alignment. - Determine the inertial measurement unit (IMU) inner gimbal angle required to align a horizon alignment mark on the CM window to the horizon at a selected time, given the spacecraft state vector, REFSMMAT, and the vehicle yaw and roll angles.

3.3.6 LM horizon alignment. - Determine the IMU outer gimbal (yaw) angle required to align the LM Z-axis in the local vertical plane at a selected time, given the spacecraft inner and middle gimbal angles. Also, determine the position of the horizon on the landing point designator (LPD) set of lines on the LM window.

3.3.7 Lift-off REFSMMAT. - Determine the lift-off REFSMMAT, given the flight azimuth and time of guidance reference release.

3.3.8 Radiation dosage. - Given a state vector and time interval of the required computation, determine the geomagnetic parameters, the radiation dose rates (REM per hour), and the cumulative radiation dose (REMS) in the CM and LM.

3.3.9 Ground track. - Determine ground track data (latitude, longitude, altitude, revolution number, azimuth, and corresponding g.e.t.). The interval at which data are output will be specified as well as the duration of the ground track data.

3.3.10 Solar activity. - Given solar flare data transmitted from the radio and optical telescopes in the Solar Particle Alert Network (SPAN), reduce the data to obtain graphs of the radio frequency burst profile and particle density as a function of time in the vicinity of the earth-moon system.

3.3.11 Spacecraft-to-sun alignment. - Determine the CSM attitude so the liquid waste dump nozzle, the electrical power system radiator, and the environmental control system radiators receive optimum heating from the sun.

3.3.12 LM docked alignment. - Determine the LM REFSMMAT or the LM gimbal angles in the docked configuration.

3.3.13 CSM attitude to perform the docking maneuver. - Given the CSM and LM REFSMMAT and the instrument unit (IU) gimbal angles for the docking attitude, determine the CSM attitude for the docking maneuver following the CSM/S-IVB separation and transposition.

3.3.14 Attitude for preferred REFSMMAT. - Given a vehicle REFSMMAT and a preferred REFSMMAT, determine the gimbal angles required for the vehicle REFSMMAT that would define 0, 0, 0 gimbal angles for the preferred REFSMMAT.

3.3.15 LM gimbal angles to flight director attitude indicator (FDAI) angles conversion. - Given a set of either LM gimbal angles or FDAI angles, determine the corresponding FDAI angles or LM gimbal angles.

3.3.16 Earth light illuminance. - Determine the amount of reflected earth-light in lumens per square foot on the scanning telescope of the CM.

3.3.17 Vehicle gimbal angle conversions. - Given the lift-off gimbal angles of the IU, the CM IMU, and the LM IMU, determine the three sets of gimbal angles, at a later time, prior to transposition and docking, given any one of the three sets of gimbal angles.

3.4 Orbital Maneuver Requirements

3.4.1 Navigation vector update evaluation. - Given an RTCC tracking vector and a spacecraft telemetry vector prior to a maneuver, apply the given maneuver to the telemetry vector using command module computer logic or lunar guidance computer logic, and then apply the resulting accelerations to the tracking vector. The resulting detailed maneuver tables (DMT) will be used to determine if a maneuver may be performed without a navigation vector update.

3.4.2 Maneuver evaluation. - Given a state vector before and after an orbital maneuver, a REFSMMAT, the roll angle at ignition, and the ignition time, determine the actual external-ΔV components and spacecraft attitude resulting from the maneuver.

3.4.3 Rendezvous and general orbital maneuvers. - Determine the series of maneuvers required to accomplish a rendezvous plan and generate the mission plan table, detailed maneuver table, and the rendezvous evaluation table. Also, generate any other orbital maneuvers required to successfully complete the mission.

3.5 Command Load Requirements

3.5.1 Command module computer uplink data. - Given a set of data in engineering units to be uplinked to the CMC, determine the octal equivalent of these data in the format and scaling acceptable to the CMC. Conversion of the following sets of data will be required: navigation vector update, REFSMMAT, orbital external-ΔV data, and deorbit external-ΔV data.

3.5.2 Lunar module computer uplink data. - Given a set of data in engineering units to be uplinked to the LGC, determine the octal equivalent of these data in the format and scaling acceptable to the LGC. Conversion of the following sets of data will be required: navigation vector update, REFSMMAT, and orbital external-ΔV data.

3.5.3 Engineering units to octal conversion. - Convert a number in engineering units to octal, given the scale factor, precision, and multiplier to be used.

3.5.4 Octal to engineering units conversion. - Given an octal number with its associated scale factor and precision, determine the equivalent number in engineering units.

3.5.5 Navigation vector update. - Given a state vector and a time to perform a navigation vector update, determine and output, in engineering units and the correct octal format, a navigation vector update for either the CMC, LGC, AGS, or the S-IVB onboard computers.

3.6 Optical Sighting and Antenna Pointing Requirements

3.6.1 CMC IMU alignment. - Determine the CMC REFSMMAT, given two stars, their location in the telescope or sextant field of view, and the corresponding spacecraft IMU gimbal angles.

3.6.2 LGC IMU alignment. - Determine the LGC REFSMMAT, given two stars, their location in either the alignment optical telescope (AOT) field of view or in the crew optical alignment sight (COAS) field of view, and the corresponding spacecraft IMU gimbal angles.

3.6.3 CSM star finding. - Given the current CMC IMU alignment, the current spacecraft IMU gimbal angles, and a search interval, locate up to 10 stars that will be in the telescope field of view and the position of these stars with respect to the telescope radicle pattern as well as the star acquisition of sight (AOS) and loss of sight (LOS).

3.6.4 CSM star location. - Determine the sextant shaft and trunnion angles required to center two stars in the sextant field of view, given the star identification, the current REFSMMAT, and the spacecraft IMU gimbal angles at a specified time.

3.6.5 LM star finding. - Given the current LGC IMU alignment, the vehicle REFSMMAT, and a search interval, find up to 10 stars that will be in the LM optics field of view during the search interval as well as the AOS and LOS of each star. The optics used, AOT or COAS, and their detent position or axis orientation must also be defined.

3.6.6 LM star location (AOT). - Determine the shaft and trunnion angles required to center two stars in the AOT field of view, given the star identifications, the current REFSMMAT, the detent position, and the spacecraft IMU gimbal angles at a specified time.

3.6.7 LM star location (COAS). - Determine the star X-position and star elevation angle to center two stars in the COAS field of view, given the star identifications, the current REFSMMAT, the axis mount of the COAS, and the spacecraft IMU gimbal angles at a specified time.

3.6.8 Point AOT with CSM. - Given the CMC REFSMMAT, the docking angle, and an RTCC state vector, determine the CMC gimbal angles required to center a specified star in the AOT field of view at a specified time. The detent position of the AOT must also be specified.

3.6.9 Steerable antenna data. - Given the spacecraft attitude, REFSMMAT, and time of sighting, determine the pitch and yaw angles of any one of the three onboard antennas necessary to point the antenna at a specified earth target. Alternately, given the pitch and yaw angles of the antenna, determine the spacecraft attitude necessary to point the antenna at the selected target.

3.6.10 Ground target sighting. - Determine the spacecraft IMU gimbal angles, time of arrival at the desired line of sight to the target, and the time and central angle of closest approach, given the target location, REFSMMAT, and desired sextant configuration.

3.6.11 Celestial target sighting. - Given a celestial target location, REFSMMAT, and fixed sextant configuration, determine the spacecraft IMU gimbal angles required to center the target in the sextant field of view. Also, compute the central angle and time of closest approach, the time of arrival at the line of sight to the target, and the earliest point at which the line of sight does not pass through the atmosphere of the earth.

3.7 Flight Planning and Experiments Work Schedule Requirements

3.7.1 Radar data. - Given a state vector, minimum elevation angle, and any maneuvers to be performed during a specified time interval, determine the following quantities for specified radar sites: spacecraft acquisition and loss time, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and the duration of the pass.

3.7.2 Spacecraft daylight-darkness. - Given a state vector, time interval to be considered, and any maneuvers to be performed during the interval, determine the time and spacecraft position of sunrise, sunset, and terminator crossing.

3.7.3 Spacecraft moon sighting. - Given a state vector, time interval to be considered, and any maneuvers to be performed during the interval, determine the time and spacecraft position of moonrise and moonset.

3.7.4 Computed events. - Determine the orbital events (apogee, perigee, ascending node, and revolution number) and related times of these events, given an initial state vector, time interval to be considered, and any maneuvers performed during the interval.

3.7.5 Landmark sighting. - Given the landmark number, time interval to perform the landmark search, and a state vector, determine the following quantities: the spacecraft acquisition and loss time, slant range and azimuth at acquisition, minimum slant range, maximum elevation angle, and duration of the pass.

3.7.6 Spacecraft star sighting. - Given a star identification number, revolution number, and a state vector, determine the times of star rise and star set relative to the spacecraft and the time and position in which the star-earth-spacecraft central angle is a minimum (closest approach).

3.7.7 Closest approach. - Determine the spacecraft closest approach to a specified ground target, given target identification, revolution number, and a state vector.

3.7.8 Pointing data. - Given a state vector, target identification, REFSMMAT, and time interval to perform the target search, determine the spacecraft to target look angles (gimbal angles and local vertical/local horizontal angles), and the target to spacecraft look angles (elevation angle and azimuth angle). Also, compute the spacecraft acquisition and loss times, maximum elevation angle, minimum slant range, altitude, and elapsed time of the pass.

3.8 Systems Requirements

3.8.1 Mass properties and aerodynamics. - Given the weights, center of gravity, moments of inertia of the consumables tanks, and any miscellaneous items to be considered, determine the following quantities:

a. Aerodynamics for CM entry

b. Center of gravity locations for different CM, SM, and LM consumable and equipment configurations

c. Mass properties table for a specific oxidizer to fuel mixture ratio for the CSM and LM (docked and APS alone). The mass properties table will be determined for these configurations for both LM active and CSM active configurations.

d. Digital autopilot command load for a specific SPS thrust level

3.8.2 SM RCS propellant profile. - Determine the complete SM RCS propellant budget, given the spacecraft mass properties, the control mode for each maneuver, the RCS jet selection, and a timeline of maneuvers.

3.8.3 SM RCS propellant status. - Determine the current SM RCS propellant available using the primary or auxiliary system, given the quad selection, the corresponding helium pressures and temperatures, the tank expulsion efficiencies, and the RCS oxidizer to fuel mixture ratios.

3.8.4 Total heat load during extravehicular mobility unit use. - Determine total heat load on the coolant loop of the extravehicular mobility unit during periods of extravehicular activity using the following methods:

a. Total heat load by metabolic method

b. Total heat load by loop equation method

c. Total heat load by fuel H_2O pressure method

d. Total heat load by H_2O usage rates

3.8.5 LM RCS propellant budget. - Determine the LM RCS propellant consumables budget based on an input mission timeline event description.

3.8.6 DPS supercritical helium pressure profile. - Determine the pressure profile of the DPS supercritical helium fuel pressure system for each DPS maneuver. The profile will include the maximum helium pressure, the helium pressure at the end of a burn, the helium mass remaining at the end of a burn, and a tabulation of pressure versus time during the coast period prior to the maneuver, during the maneuver, and for a period of time following the maneuver.

3.8.7 LM telemetry diagnostics. - Determine the common failure points in the telemetry downlink system, given the code number of the failed telemetry points.

3.8.8 LM electrical power system analysis. - Given the premission LM power distribution network parameters (on magnetic tape), the mission timeline, and any new component configuration, determine the capability of the LM electrical power system to support the mission.

3.9 Deorbit Requirements

3.9.1 Primary landing area (PLA). - Determine the deorbit ignition time, IMU gimbal angles at ignition, and the time of reverse the bank angle from the initial bank angle in order to achieve a target latitude and longitude in one of the primary landing areas. The maneuver will be based on a specified attitude, REFSMMAT,* and an incremental velocity or a velocity and flight-path angle constraint at entry interface. The entry profile will consist of a lift vector orientation from 400, 000 feet to a specified g-level, followed by a bank-reverse-bank angle entry to drogue chute deployment.

3.9.2 Contingency landing area (CLA). - Determine the deorbit ignition time and IMU gimbal angles at ignition to achieve a target longitude in one of the contingency landing areas based on a specified attitude, REFSMMAT,* an incremental velocity or a velocity and flight-path angle constraint at entry interface, and an entry profile. This profile will consist of: (1) a lift vector orientation from 400, 000 feet to a specified g-level, followed by a constant bank angle to drogue chute deployment, or (2) a rolling entry from 400, 000 feet to drogue chute deployment.

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*The REFSMMAT will either be given or computed from the spacecraft body attitude and IMU gimbal angles.

3.9.3 Command module RCS deorbit. - Determine the CM RCS deorbit ignition time and IMU gimbal angles at ignition to achieve a target longitude based on an incremental velocity, spacecraft attitude, REFSMMAT,* and a rolling entry from 400, 000 feet to drogue chute deployment.

3.9.4 Hybrid deorbit. - Determine the SM and CM RCS deorbit ignition times in order to achieve a target longitude, given the incremental velocities and spacecraft attitudes for each maneuver, REFSMMAT,* and an entry profile consisting of a lift vector orientation to a specified g-level, followed by a zero-lift entry to drogue chute deployment. Also determine the landing point based on the actual hybrid deorbit performed. The SM RCS maneuver will be considered as having been performed nominally, and the actual incremental velocities will be used to define the CM RCS maneuver. The entry profile will be the same as that mentioned previously.

3.9.5 Hybrid deorbit without SLA separation. - Given a state vector before S-IVB/CSM separation, the time of initiation of the LOX dump maneuver, and the CMC REFSMMAT,* determine the CM RCS deorbit, delta velocity at apogee, and the IMU gimbal angles at ignition to achieve a target longitude. The entry profile will nominally consist of a lift vector down to a 1-g deceleration followed by a rolling entry to drogue chute deployment.

3.9.6 Deorbit with separation maneuver. - Given a state vector prior to separation, a specified separation and deorbit maneuver, and an entry profile, determine the time to perform the SPS deorbit maneuver and all the entry quantities associated with the deorbit. The separation maneuver will either occur at a fixed g.e.t. or at a fixed time interval prior to SPS ignition. The separation maneuver will be defined by the velocity increment of the burn, the spacecraft attitude, and the ignition time which will either be given (fixed time) or will be computed (fixed delta t). The deorbit maneuver will be specified in the same manner as a PLA or CLA deorbit.

3.9.7 Block data. - Determine PLA and CLA deorbit ignition times for a series of deorbits, given a velocity and flight-path angle constraint at entry interface, the spacecraft attitude at ignition, the entry profile to be flown, and the type of landing area to be considered. The deorbit data are to be computed in blocks of six revolutions and are to reflect any planned maneuvers occurring in this period. The first set of block data should reflect the S-IVB/CSM separation maneuver. For all deorbits occurring prior to the nominal S-IVB/CSM separation time, the deorbits will be computed with the separation maneuver occurring at a fixed interval of time before the deorbit ignition. All deorbits in the first block of data occurring after nominal separation time will simulate the separation maneuver at the fixed nominal time.

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*The REFSMMAT will either be given or computed from the spacecraft body attitude and IMU gimbal angles.

3.9.8 <u>Guided entry and backup guidance quantities</u>. - Determine entry monitoring system (EMS) initialization quantities, the guided entry profile, and backup guidance quantities required to reach a target landing point, given a state vector at entry interface. The state vector at entry interface will be generated by a deorbit processor that will either simulate the deorbit maneuver, given a preburn state vector, or will coast a deorbit postburn vector to entry interface.

4. RTACF PROCESSORS FOR THE APOLLO 9 MISSION

4.1 Introduction

This section of the Flight Annex presents a description of the processors that will be used in the RTACF to satisfy the Apollo 9 mission requirements that have been received to date.

4.2 Types of RTACF Output Available

There are two types of output available for the Apollo 9 processors to be used in the RTACF. The first type of output is in the form of a summary sheet which is designated as the on-line computer output, while the second type is printed off-line from a computer magnetic tape written by the processor. The summary sheets, which are output immediately after the termination of the computation, display at least the minimum number of output parameters necessary to satisfy the requirements. The off-line computer printout can be obtained, if desired, and contains additional information to supplement the on-line summary sheets.

There are presently 40 different on-line summary sheets available in the RTACF. The standard summary sheet will be used to output the results of 7 of the RTACF processors, and the FDO detailed maneuver table will be used for an additional 2 of the processors. The standard summary sheet has been formatted to include selected outputs from the 7 processors. Even though the outputs of these processors are somewhat similar, no single processor is capable of providing all the data indicated on the standard summary sheet. Thus, in order to interpret the standard summary sheet correctly, some detail of the outputs of each of the particular processors must be known. However, for any processor using the standard summary sheet, all of the required data for that processor will be included.

The remaining 38 summary sheets are formulated specifically for particular processors or functions. The details of the processors associated with each summary sheet are described in the remainder of this section.

4.3 Mode I Launch Abort Processor

The Mode I abort region is defined from lift-off until the launch escape tower (LET) jettison which occurs 15 seconds after S-IVB ignition. The Mode I landing area is completely contained in the Atlantic Continuous Recovery Area (ACRA), which extends downrange 3200 nautical miles along the flight azimuth.

Mode I abort processor. - The actual wind profiles encountered from 24 to 1 hour prior to launch are employed in this processor to predict the spacecraft landing points should a Mode I abort be performed during the first 90 seconds of the flight. Selected abort times are considered in this time interval to determine which aborts will result in a land impact about the launch area.

Inputs required:

a. Wind profile from the Kennedy Space Center

b. CM entry weight

Outputs required:

The outputs required for this processor will be displayed on the Mode I launch-abort summary sheet shown in Figure 2.

4.4 General Orbit Phase Processors

There are presently 18 processors that can be included under the classification of general orbit phase processors. From these processors, a variety of information is obtained concerning the mission during the coast periods. Such data as orbital lifetime, relative motion, groundtrack, etc., are obtained exclusive of any orbital maneuvers, optical sightings, and command load computations.

4.4.1 Orbital lifetime. - This processor computes the predicted orbital lifetime of the spacecraft, given a state vector, the aerodynamic characteristics of the vehicle, and the model atmosphere to be used.

Inputs required:

a. State vector

b. Year, month, and day of launch

c. Vehicle drag coefficient and reference area

d. Atmosphere model to be used

e. Vehicle weight

Outputs required:

Lifetime in days, hours, and minutes from lift-off and from the time of the vector.

4.4.2 K-factor. - This processor computes the atmospheric density K-factor to be used in the RTCC. The K-factor value is determined by propagating one input state vector to the time of one or more succeeding state vectors. The value of the atmospheric density multiplier is adjusted until the propagated vector and succeeding state vectors agree to some specified accuracy.

Inputs required:

a. Two or more state vectors for the same vehicle

b. Spacecraft weight

c. Spacecraft drag coefficient and reference area

Outputs required:

a. Value of K-factor

b. Probable error in K-factor

4.4.3 FDO orbit digitals. - This processor takes a state vector and computes the orbital quantities included in the RTCC FDO orbit digitals display and presents them in a similar format.

Inputs required:

a. RTCC state vector

b. Spacecraft weight

c. Threshold time or revolution number

Outputs required:

The outputs required for this processor are displayed on the FDO orbit digitals summary sheet shown in Figure 3.

4.4.4 Relative motion. - The relative motion processor computes the relative motion quantities associated with an active and passive vehicle. The quantities are measured relative to a curvilinear coordinate system whose origin can be centered at either vehicle.

Inputs required:

a. RTCC state vectors for the two vehicles

b. Current weight of each vehicle

c. Current drag coefficient and reference area for each vehicle

d. Vehicle at which coordinate system will be centered

e. Any maneuvers performed by the active vehicle

Outputs required:

The outputs required for this processor are displayed on the relative motion summary sheet shown in Figure 4.

4.4.5 CSM IMU horizon alignment. - This processor will calculate the IMU inner gimbal angle required to align a horizon alignment mark on the CM window to the horizon at a specified time.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Horizon monitor attitude

d. Time of computation

e. Spacecraft weight

f. IMU roll and yaw gimbal angles

Outputs required:

IMU inner gimbal angle at the specified time

4.4.6 LM IMU horizon alignment. - This processor will accept the inner and middle LM IMU gimbal angles and compute the outer gimbal angle required to align the Z-body axis of the LM in the local vertical plane. Also computed is the position of the horizon on the landing point designator (LPD).

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Inner and middle LM IMU gimbal angles

d. Time for alignment

Outputs required:

a. Yaw (outer) gimbal angle

b. Position of horizon on the LPD

4.4.7 Lift-off REFSMMAT. - This processor computes the REFSMMAT that will be used from lift-off until the IMU is realigned in orbit.

Inputs required:

a. Flight azimuth

b. Time of guidance reference release

Outputs required:

Lift-off REFSMMAT

4.4.8 Radiation evaluation. - This processor takes a state vector and, at given intervals along a trajectory, determines geomagnetic parameters and the radiation dose rates in both the command module and in the lunar module. It also calculates total REM dose in both modules by integrating the radiation dose rates over a particular portion of the trajectory.

Inputs required:

a. RTCC state vector

b. Spacecraft weight

c. Spacecraft drag coefficient and reference area

Outputs required:

The outputs required for this processor will be displayed on the radiation evaluation summary sheet shown in Figure 5.

4.4.9 Orbital data for the Public Affairs Office (PAO). - The PAO requirements are not satisfied by any single processor. The outputs satisfying the desired request will be displayed on the summary sheet of the processor used.

4.4.10 Ground track. - The ground track requirements are not satisfied by any single processor. Ground track data capability is available with any General Electric Missile and Satellite Simulation Multivehicle (GEMMV) or Apollo Reference Mission (ARMACR) processor by selection of the ground track option. The output satisfying this requirement will be displayed on the ground track summary sheet shown in Figure 6.

4.4.11 Solar Particle Alert Network (SPAN). - The SPAN processor will process solar flare data received from the optical and radio telescopes in the SPAN. The input data will be received in the form of a punched paper tape which is generated at the space environment console. The SPAN processor will first transfer these data from the paper tape to a magnetic tape. It will then process the magnetic tape and display the resulting data in a form from which the radiation hazard in the earth's vicinity can be determined.

4.4.12 Spacecraft-to-sun alignment. - This processor computes the CSM attitude in gimbal angles necessary to expose certain areas of the vehicle to direct sunlight for heating purposes. These areas include the liquid waste dump nozzle and the electrical power and environmental control system radiators.

Inputs required:

a. Area of the spacecraft to be exposed to the sun

b. REFSMMAT

c. Right ascension and declination of the sun

Outputs required:

IMU gimbal angles required to orient the specified areas to the sun

4.4.13 LM and CSM docking alignment. - This processor will be used to compute either the LM REFSMMAT or the LM attitude for the docked CSM/LM configuration, given the CSM REFSMMAT and attitude and the docking angle between the LM and CSM. Alternately, it will accept both the LM REFSMMAT and attitude and compute the CSM gimbal angles necessary for docking.

The processor has three options: option 1 computes the LM REFSMMAT, given the LM attitude; option 2 computes the LM attitude (gimbal angles and FDAI angles), given the LM REFSMMAT; and option 3 computes the CSM attitude (gimbal angles), given the LM attitude and REFSMMAT and the CSM REFSMMAT.

Inputs required:

Option 1:

a. CSM REFSMMAT

b. CSM gimbal angles

c. Docking angle

d. LM gimbal angles or FDAI angles

Option 2:

a. CSM REFSMMAT

b. LM REFSMMAT

c. CSM gimbal angles

d. Docking angle

Option 3:

a. CSM REFSMMAT

b. LM REFSMMAT

c. LM gimbal angles or FDAI angles

d. Docking angle

Outputs required:

The outputs required for this processor are displayed on the docking alignment processor summary sheet shown in Figure 7.

4.4.14 Attitude for preferred REFSMMAT. - This processor will be used to define the spacecraft attitude (gimbal angles and FDAI angles) so that if the spacecraft had these gimbal angles and the current REFSMMAT and then set all gimbal angles to zero (maintaining its inertial attitude), it would have the preferred REFSMMAT.

In practice, the processor accepts the current and desired REFSMMATS, computes the gimbal angles for the current REFSMMAT which define 0, 0, 0 gimbal angles for the preferred REFSMMAT, and outputs a set of gimbal angles and FDAI angles which used in conjunction with the current REFSMMAT define the spacecraft's attitude necessary to switch to the preferred REFSMMAT and read 0, 0, 0 gimbal angles.

Inputs required:

a. Current REFSMMAT

b. Preferred REFSMMAT

Outputs required:

Gimbal angles and FDAI angles for use with the current REFSMMAT

4.4.15 LM gimbal angles to FDAI angles conversion. - This processor will be used to convert LM gimbal angles to FDAI angles or vice versa.

Inputs required:

LM gimbal angles or FDAI angles

Outputs required:

LM gimbal angles or FDAI angles

4.4.16 Earth light illuminance. - This processor accepts the spacecraft attitude and shaft angle of the scanning telescope and calculates the amount of reflected earth light (in lumens/square feet) on the scanning telescope. If sunlight falls on the telescope, an appropriate message is output in addition to the computed earth light.

This is a post processor to ARMACR. The spacecraft gimbal angles and the REFSMMAT are passed via the 200-word record written by ARMACR.

Inputs required:

Shaft angle of the scanning telescope

Outputs required:

Illuminance on the scanning telescope

4.4.17 Open hatch thermal control. - Given an RTCC state vector and a CSM REFSMMAT, this processor will compute the IMU gimbal angles necessary for the CSM to align itself so that optimum lighting is available for the extravehicular activity.

In practice, the alignment is defined as: -Z axis coincident with the sun vector and +X axis pointed toward the South Celestial Pole. The vehicle is then pitched down 15 degrees and rolled left 80 degrees. This attitude gives acceptable lighting on the open hatch while EVA is being performed, yet does not permit direct exposure of the sun into the spacecraft through the hatch.

Inputs required:

a. RTCC state vector

b. REFSMMAT

Outputs required:

The outputs required are displayed on the EVA open hatch thermal control summary sheet shown in Figure 8.

4.4.18 Gimbal angle conversion. - Prior to separation of the CSM from the S-IVB/LM combination, this program takes a set of either S-IVB, CSM, or LM gimbal angles and outputs the remaining two sets based on S-IVB, CSM, and LM lift-off gimbal angles.

Inputs required:

a. S-IVB gimbal angles at lift-off

b. CSM gimbal angles at lift-off

c. LM gimbal angles at lift-off

d. Any one of the three sets at a later time

Outputs required:

Three sets of gimbal angles (S-IVB, CSM, and LM)

4.5 Orbit Maneuver Processors

The orbit maneuver requirements are satisfied by three ARMACR processors and a general rendezvous support program. Two of the ARMACR processors are used to evaluate orbital maneuvers: one determines the effect of any errors in the CMC state vector on an upcoming maneuver while the other evaluates the results of a completed maneuver. The third processor can simulate a variety of maneuvers once the maneuver data has been specified. The rendezvous program is used to plan the rendezvous maneuvers and can be used to compute any other maneuvers performed during the Apollo 9 mission.

4.5.1 Navigation vector update evaluation. - This processor determines whether a navigation vector update is necessary prior to a planned maneuver. A spacecraft telemetry vector and an RTCC state vector are input and propagated to the maneuver time. The maneuver is applied to the spacecraft telemetry vector, and the acceleration profile resulting from the use of the CMC or LGC guidance logic is applied to the RTCC tracking vector. After the completion of the maneuver, the vectors are compared to see if a navigation vector update is required.

Inputs required:

a. RTCC state vector

b. Spacecraft telemetry vector

c. REFSMMAT

d. Maneuver targets

e. Guidance mode

f. Spacecraft weight

Outputs required:

The outputs required for this processor are displayed on the FDO detailed maneuver table shown in Figure 9.

4.5.2 General orbit maneuver. - This processor can simulate any orbital maneuver given the appropriate burn quantities, spacecraft attitude during the maneuver, inertial platform alignment, and the type of guidance mode to be used during the maneuver.

Inputs required:

a. RTCC state vector

b. Ignition time

c. Burn duration or incremental velocity

d. Attitude at ignition

e. REFSMMAT or IMU gimbal angles

f. Vehicle configuration (docked/undocked - CSM active/LM active)

Outputs required:

The outputs required for this processor are displayed on the FDO detailed maneuver table shown in Figure 9.

4.5.3 Maneuver evaluation. - This processor computes a maneuver that is equivalent to the maneuver actually performed. A preburn vector is propagated to the time of the postburn vector; both vectors are then propagated back to the maneuver ignition time. At this point, the actual spacecraft attitude and external delta V components are computed.

Inputs required:

a. Preburn state vector

b. Postburn state vector

c. Time of ignition

d. REFSMMAT or IMU gimbal angles

e. Roll angle at ignition

f. Vehicle configuration (docked/undocked - CSM active/LM active)

Outputs required:

The outputs required for this processor are displayed on the maneuver evaluation summary sheet shown in Figure 10.

4.5.4 Apollo Real-Time Rendezvous Support Program. - The Apollo Real-Time Rendezvous Support (ARRS) Program was designed to provide a dual purpose tool for both mission planning and rendezvous mission support in the RTACF. ARRS is composed of a number of processors and routines required to support a rendezvous mission. The processors and routines that will be of concern to the Apollo 9 mission are described below:

a. The general purpose maneuver processor (GPMP) is used to compute impulsive maneuvers at a specified point in an orbit to achieve desired orbital conditions.

b. The two-impulse and terminal phase processor computes a set of two impulsive maneuvers by specifying when they should be performed and by specifying the conditions, such as phase and height offsets, at the final maneuver point.

c. The mission plan table (MPT) processor accepts vectors before and after impulsive maneuvers and computes the required finite burn quantities necessary to achieve the orbit after the maneuver.

d. The relative print routine computes relative quantities, such as range, range rate, and look angles, between two orbiting vehicles.

e. The tracking routine computes the tracking station coverage of a vehicle from its initial vector through all the maneuvers that have been established in the mission plan table.

f. The concentric rendezvous processor computes a rendezvous plan by using concentric flight-plan logic.

g. The vector conversion routine converts vectors from one coordinate system to a number of other coordinate systems.

h. The trajectory update routine accepts an RTCC state vector, executes the maneuvers in the MPT, and computes the resulting trajectory.

The inputs required by the ARRS Program are described in Reference 7.

4.6 Command Load Processors

The RTACF possesses the capability of generating command loads in octal with the proper scaling and format to be directly uplinked to the CMC or LGC. The capability also exists for receiving certain down-linked quantities from the CMC or LGC and converting them to the appropriate engineering units. A program was developed to perform these special conversions, as well as any number of general conversions from engineering units to octal or vice versa. In addition, a processor was developed to generate CMC, LGC, AGS, or S-IVB state vector updates at a given time from an RTCC tracking vector.

4.6.1 Command formatting and conversion. - This program contains seven options in which data in engineering units are converted to octal format or data in octal are converted to engineering units. Six of the options are concerned with up- or down-linked CMC or LGC quantities and possess preset formats, scale factors, and octal precisions. The seventh option is for general conversion from engineering units to octal, or vice versa, given the number, scale factor, precision, and any multipliers.

To simplify discussions of the seven options, only conversions in one direction will be considered. Each option, however, also contains the capability to convert in the other direction.

4.6.1.1 Navigation vector update: The navigation vector update option converts a state vector in the Besselian coordinate system with the units of feet and feet per second to octal units acceptable to the CMC or LGC.

Inputs required:

a. Position components of a state vector in feet

b. Velocity components of a state vector in feet per second

c. Vector time in g. e. t.

Outputs required:

The outputs required from this option are displayed on the command load navigation update summary sheet shown in Figure 11.

4.6.1.2 Orbital external ΔV: This option converts the target external ΔV components for an orbital maneuver, the maneuver time, and the weight prior to the maneuver to octal units to be uplinked to the CMC or LGC.

Inputs required:

a. Maneuver time in g. e. t.

b. External ΔV components in feet per second

c. Maneuver weight in pounds

Outputs required:

The outputs required from this option are displayed on the orbital external ΔV summary sheet shown in Figure 12.

4.6.1.3 Deorbit external ΔV: This option converts the target deorbit external ΔV components, the deorbit ignition time, the weight at deorbit, and the target point to the octal format required for CMC uplink.

Inputs required:

a. Maneuver time in g. e. t.

b. External ΔV components in feet per second

c. Maneuver weight in pounds

d. Latitude and longitude of the target in degrees

Outputs required:

The outputs required from this option are displayed on the deorbit external ΔV summary sheet shown in Figure 13.

4.6.1.4 REFSMMAT update: This option converts a REFSMMAT to the octal format required for a CMC or LGC update.

Inputs required:

Elements of REFSMMAT

Outputs required:

The outputs required from this option are displayed on the REFSMMAT update summary sheet shown in Figure 14.

4.6.1.5 Numeric RTCC restart: This option converts a spacecraft vector in the CMC numeric units to an RTCC vector in engineering units. This conversion routine contains the transformation from the Besselian to the Greenwich inertial coordinate system.

Inputs required:

a. Position components of spacecraft vector in numeric units

b. Velocity components of spacecraft vector in numeric units

c. Vector time in numeric units

Outputs required:

The outputs required from this option are displayed on the command load navigation update summary sheet shown in Figure 11.

4.6.1.6 Alphanumeric RTCC restart: This option is similar to option 5 except that the inputs are in the CMC alphanumeric units.

Inputs required:

Same as option 5 with inputs in alphanumeric units

Outputs required:

Same as option 5

4.6.1.7 General octal conversion: This option converts any number from engineering units to octal after the scale factor and the octal precision have been specified. The option also has the capability of converting the number from one set of engineering units to another set by specifying a multiplier.

Inputs required:

a. Number to be converted

b. Scale factor

c. Octal precision

d. Multiplier

Outputs required:

The outputs required from this option are displayed on the general octal conversion summary sheet shown in Figure 15.

4.6.2 Navigation vector update. - This ARMACR processor takes an RTCC state vector, propagates it forward to the navigation vector update time, and outputs the state vector at that time in engineering units and either the CMC, LGC, or S-IVB octal format. In addition, the processor will update a vector and output it in engineering units only for the AGS.

Inputs required:

a. RTCC state vector

b. Time of navigation vector update

c. Spacecraft weight, drag coefficient, and reference area

Outputs required:

The outputs required for this processor will be displayed on either the CMC or S-IVB navigation vector update summary sheet shown in Figures 11 and 16, respectively. The AGS navigation update is shown in Figure 17.

4.7 Optical Sighting and Antenna Pointing Processors

The optical equipment aboard the CM consists of the following instruments: a scanning telescope, a sextant, and a boresight. The scanning telescope and sextant are interconnected with equipment common to the inertial and computer subsystems to form the primary onboard optical navigational subsystem. The scanning telescope is a single line-of-sight, unit power, wide field instrument. It is used for landmark tracking, in conjunction with IMU attitude reference, as an acquisition instrument for the sextant, for backup alignment of the inertial attitude sensors, and as a general viewing instrument. The sextant is a highly accurate, dual line-of-sight, 28-power telescope with a 1.8-degree field of view. It is used primarily to determine star to landmark or horizon angle navigation measurements, to compute star measurements for IMU alignment, and as a high power general viewing instrument.

The boresight is not part of the optical navigation subsystem and has no coupling with either the inertial or computer subsystems. It is used as a general viewing instrument and as a backup for IMU alignment if the optical subsystem should fail. It is a compact, low magnification, 5-degree field-of-view telescope whose null position line of sight is parallel to the spacecraft roll axis.

Two additional instruments make up the optical equipment onboard the LM: the alignment optical telescope (AOT) and the crewman optical alignment sight (COAS). The AOT is a unity power telescope with a 60 degree field of view and has three sighting (detent) positions; the 0 degree position is in the X-Z body plane and 45 degrees above the Z axis; the other positions are 60 degrees to the right (+60) or left (-60) of the 0 degree position. When the CSM and LM are undocked, three other positions are available. They are rear facing (grouped about the -Z axis) and spaced as previously described. The funciton of the AOT is to supplement the rendezvous radar by measuring azimuth and elevation angles to stars for alignment of the LM IMU stable member. The COAS is a 10 degree field-of-view instrument with only one degree of freedom. However, it may be placed on the commander's side in either the overhead window (X-axis mount) or the forward window (Z-axis mount).

The Apollo Generalized Optics Program (AGOP) includes the functions of the optical support table (OST) processor, the star sighting table (SST), and the lunar optical support table (LOST) processor. AGOP has options to compute steerable antenna data to support the S-band steerable antenna, the rendezvous radar antenna, and the S-band (high-gain) antenna. Other options exist that will be used in support of lunar trajectory missions but will not be used on Apollo 9. These options were verified and used operationally on Apollo 8 and are listed here only for a program overview. They are:

a. Cislunar navigation

b. Reference body locations

c. Star catalog listing

d. Passive thermal control (PTC) attitude computation

e. Terminator-horizon computation

The AGOP is not an integrating program but requires the assistance of an integrating program to generate the state vector, vehicle attitude, and other related parameters at the time of optical sightings or antenna pointing. Therefore, the AGOP is used as a post-processor to the ARMACR general purpose processor which generates the trajectory data needed for initialization.

In the program description which follows, the inputs listed for each option are those required by the AGOP with no distinction made between those input directly and those obtained from the ARMACR general purpose processor.

4.7.1 Optical support table. - The OST was developed to verify any new IMU alignment by computing the star sighting angles for the sextant, AOT, and COAS and the view-field coordinates of stars with respect to the reticle patterns of these instruments. The program contains five options that use the OST summary sheet shown in Figure 18.

4.7.1.1 IMU alignment: The REFSMMAT corresponding to a new IMU alignment is computed, given the identification of the two stars sighted, the sextant shaft and trunnion angles for each star, and the spacecraft attitude at the time of the star sightings.

Inputs required:

a. RTCC state vector

b. Spacecraft attitude in IMU gimbal angles

c. Two star identifications

d. The sextant shaft and trunnion angles of each star

e. Time of star sighting

Outputs required:

REFSMMAT associated with the new IMU alignment

4.7.1.2 LM IMU alignment: The REFSMMAT corresponding to a new LGC IMU alignment may be computed based on either one of two optical instruments onboard the LM. They are the alignment optical telescope and the crew optical alignment sighting instrument.

Alignment optical telescope: Given the detent position of the telescope, the spacecraft attitude, two star identifications, and the AOT shaft and trunnion angles for each star, the corresponding REFSMMAT may be computed.

Inputs required:

a. RTCC state vector

b. Spacecraft attitude (LM IMU gimbal angles or FDAI angles)

c. AOT detent position (0, +60, and -60 degrees)

d. Time of the two star sightings

e. AOT shaft and trunnion angles of each star

f. Two star identifications

Outputs required:

REFSMMAT associated with the new alignment

Crewman optical alignment sighting: Given the spacecraft axis along which the COAS is mounted, the spacecraft attitude, the reticle elevation angles, and X-positions of two stars, the REFSMMAT may be computed.

Inputs required:

a. RTCC state vector

b. Spacecraft attitude (LM IMU gimbal angles or FDAI angles)

c. Spacecraft axis along which the COAS is mounted

d. COAS reticle elevation angle for each star

e. COAS reticle X-position angle for each star

f. Time of the two star sightings

g. Two star identifications

Outputs required:

REFSMMAT associated with the new alignment

4.7.1.3 CM star finding: The star finding option locates up to 10 stars which are in the scanning telescope field of view over a specified time interval at a specified spacecraft attitude. Acquisition and loss times are determined for each star.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. IMU gimbal angles

d. Time interval to be searched

Outputs required:

Acquisition and loss times of the stars found

4. 7. 1. 4 LM star finding: Given the LM attitude, search interval, and instrument to be used for sighting, this processor locates up to 10 stars which are available in the field of view of the specified instrument over the period to be searched and determines the acquisition and loss times of each of the stars.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. IMU gimbal angles or FDAI angles

d. Instrument to be used for sighting

e. Time interval to be searched

f. Detent position if AOT is used

g. Axis along which COAS is mounted (if used)

Outputs required:

Acquisition and loss times of the stars found

4. 7. 1. 5 Point AOT with CSM: This option will accept a state vector, REFSMMAT, one star identification, and the AOT position and will compute the CSM attitude (gimbal angles) at the time of sighting.

Inputs required:

a. CSM REFSMMAT

b. RTCC state vector

c. Search interval

d. AOT detent position

e. Star identification

f. Docking angle

Outputs required:

CSM IMU gimbal angles at the time of star sighting

4. 7. 2 Star sighting table. - The SST option was developed to satisfy those optical sighting requirements that are concerned with ground and celestial target sightings. This option computes the time of arrival at a specified line of sight to the target and the spacecraft attitude or sextant orientation required to view the target.

The program contains six options, the outputs of which are displayed on the star sighting table shown in Figure 19. In the description of each option, the inputs listed are the combined inputs required for both the ARMACR general purpose processor and the SST Program option.

4.7.2.1 Ground target sighting: This option computes the required spacecraft attitude for viewing a specified ground target at a given line-of-sight elevation angle with the CM optical system in a fixed configuration.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Target identification or location

d. Elevation angle of line of sight to the target

e. Sextant shaft and trunnion angles

Outputs required:

a. IMU gimbal angles

b. Time of arrival at the desired line of sight to the ground target

c. Central angle and time of closest approach

4.7.2.2 Ground target sighting with movable sextant configuration: This option computes the required sextant shaft and trunnion angles for viewing a ground target at a fixed spacecraft attitude. The inputs and outputs are identical to those in option 1 with the exception that the spacecraft attitude is input and the sextant shaft and trunnion angles are output.

4.7.2.3 Celestial target sighting with fixed sextant configuration: This option computes the required spacecraft attitude and time for viewing a specified celestial target with the sextant in a fixed configuration.

Inputs required:

a. RTCC state vector

b. REFSMMAT

c. Target identification or location

d. Sextant shaft and trunnion angles

Outputs required:

a. IMU gimbal angles

b. Time of arrival at the line of sight

c. Minimum target-earth-spacecraft central angle (closest approach)

d. Time of closest approach

e. Right ascension and declination of the line of sight

f. Earliest subsatellite longitude at which the line of sight does not pass through the earth's atmosphere

4.7.2.4 Celestial target sighting with movable sextant configuration: This option computes the required sextant shaft and trunnion angles for viewing a celestial target at a fixed spacecraft attitude. The inputs and outputs from this option are identical to those in option 3 with the exception that the spacecraft attitude is input and the sextant shaft and trunnion angles are output.

4.7.2.5 Celestial target sighting with AOT (fixed vehicle attitude): Given the spacecraft attitude, the AOT detent position, and the time of sighting, this option computes the right ascension and declination of the center of the field of view of the AOT at the time of sighting.

Inputs required:

a. Time of sighting

b. REFSMMAT

c. RTCC state vector

d. IMU gimbal angles

e. AOT detent position

Outputs required:

a. Right ascension and declination of the center of the AOT field of view

b. Central angle

4.7.2.6 Celestial target sighting with AOT (movable vehicle attitude): Given the right ascension and declination of the target point, the AOT detent position, and time of sighting, this option computes the IMU gimbal angles of the spacecraft at the time of sighting.

Inputs required:

a. Time of sighting

b. REFSMMAT

c. RTCC state vector

d. Right ascension and declination of the celestial target point

e. AOT detent position

Outputs required:

a. IMU gimbal angles

b. Central angle

4.7.3 Antenna pointing. - There are two antennas onboard the LM: the steerable S-band and the rendezvous radar. The CSM antenna is the deep-space or high-gain (also S-band) antenna. All three antennas are gimballed in pitch and yaw. This processor will accept a state vector and spacecraft attitude (either CSM or LM) and compute the pitch and yaw angles necessary to point the specified antenna at a specified ground based target site. This computation also outputs the related azimuth and elevation angles required for the ground station to acquire the spacecraft.

An additional option provides the capability of fixing the position (pitch and yaw) of the specified antenna and computing the necessary spacecraft IMU gimbal angles for pointing the antenna at the specified ground station.

Inputs required for movable antenna option: :

a. Vehicle identification

b. Instrument identification

c. Site ID or location (longitude, geodetic latitude, and altitude)

d. Spacecraft gimbal angles

e. REFSMMAT

f. Minimum elevation angle at which the site can acquire the spacecraft

Inputs required for fixed antenna option:

a. Vehicle identification

b. Instrument identification

c. Site ID or location (longitude, geodetic latitude, and altitude)

d. Instrument attitude (pitch and yaw)

e. Minimum elevation angle at which the site can acquire the spacecraft

Outputs required for the movable antenna option:

The outputs required for this processor are displayed on the Steerable Antenna Pointing Program summary sheet shown in Figure 20.

Outputs required for the fixed antenna option:

Outputs required for the fixed antenna option are the same as for the movable antenna option except that spacecraft IMU gimbal angles are output in lieu of the antenna pitch and yaw angles.

4.8 Work Schedule Processor

The RTACF work schedule processor was developed to display, in graphical form, those mission and orbit related events that occur in a specified interval of time during the mission. The processor was intended to operate in a real-time environment and to generate a work schedule that would reflect any anomalies or alternate procedures that might develop during the mission.

The work schedule processor is divided into three separate modules. Module I employs any orbit phase ARMACR processor and is used to generate an ephemeris tape that becomes the input to the next module. The ephemeris tape contains all the pertinent orbit and maneuver data in the specified time interval. Module II processes the ephemeris tape and has the capability of generating any of the following data: radar, spacecraft daylight-darkness, spacecraft moon sighting, computed events, landmark sighting, spacecraft star sighting, closest approach, and pointing data. These data are also saved on an interface tape, which serves as the input to Module III, together with any comments to be included in the work schedule. The processor may be terminated at this point if only the results from Module II are desired. The processor output then consists of the summary sheets which are shown in Figures 22 through 28.

The execution of Module III is performed when the work schedule is desired. The module sorts the information contained on the interface tape and generates a plot tape which is converted to the work schedule format shown in Figure 29.

The data generated in Module II comprise the bulk of the information contained in the work schedule. The eight Module II options are described below, and the inputs required for execution of these options are listed. These inputs are in addition to the following Module I inputs which are used to generate the ephemeris tape.

a. RTCC state vector

b. Spacecraft weight

c. Time interval to be considered

d. Orbital maneuver timeline

4.8.1 Radar. - The radar option computes the spacecraft acquisition and loss times, maximum elevation, and slant range for all spacecraft passes over network stations for some specified interval of time.

Inputs required:

a. Network stations desired

b. Minimum acceptable elevation angle

c. Time interval in which radar data is needed

Outputs required:

The output quantities from this option are displayed on the radar summary sheet shown in Figure 21.

4.8.2 Daylight-darkness. - The daylight-darkness option computes the time and position, sun azimuth, and local vertical/local horizontal (LV/LH) pitch and yaw angles at which the spacecraft enters and leaves the earth's shadow and the longitude, geodetic latitude, and altitude of terminator rise and set.

Inputs required:

Time interval in which daylight-darkness data are required

Outputs required:

The output quantities from this option are displayed on the daylight-darkness summary sheet shown in Figure 22.

4.8.3 Moon sighting. - The moon sighting option computes the time and position in which the moon is visible from the spacecraft.

Inputs required:

Time interval in which moon sightings are required

Outputs required:

The output quantities from this option are displayed on the moon sighting summary sheet shown in Figure 23.

4.8.4 Computed events. - The computed events option computes the time, position, and altitude of apogee and perigee in addition to the time and location of the ascending node and the Cape crossing time.

Inputs required:

Time interval in which computed events are required

Outputs required:

The output quantities from this option are displayed on the computed events summary sheet shown in Figure 24.

4.8.5 Landmark sighting. - The landmark sighting option computes the spacecraft acquisition and loss times, maximum elevation, and slant range for all spacecraft passes over the specified landmarks during the requested interval of time.

Inputs required:

a. Landmark number

b. Time interval during which landmark sightings are required

Outputs required:

The output quantities from this option are displayed on the landmark sighting summary sheet shown in Figure 25.

4.8.6 Star sighting. - The star sighting option determines the time interval during which a star is visible from the spacecraft and the closest approach (minimum star-earth-spacecraft central angle) data associated with the star sighting in the requested time interval. The closest approach data include the time, central angle, and spacecraft attitude at the closest approach.

Inputs required:

a. Star identification

b. Time interval during which star sightings are required

Outputs required:

The output quantities from this option are displayed on the star sighting summary sheet shown in Figure 26.

4.8.7 Closest approach. - The closest approach option computes the time, position above the earth, and altitude of the spacecraft at its closest approach to a ground target.

Inputs required:

a. Target identification

b. Time interval during which these data are required

Outputs required:

The output quantities from this option are displayed on the closest approach summary sheet shown in Figure 27.

4.8.8 Pointing data. - The pointing data option computes the spacecraft-to-target and target-to-spacecraft look angles, as well as the normal radar tracking data.

Inputs required:

a. Target identification

b. Time interval during which pointing data are required

c. REFSMMAT

Outputs required:

The output quantities from this option are displayed on the pointing data summary sheet shown in Figure 28.

4.9 Systems Programs

The systems programs were designed to keep a record of the amount of consumables used during the mission and to update the CSM and LM mass properties to reflect this consumption as well as the result of vehicle reconfigurations. Eight programs are employed in the RTACF to generate these data. One program computes the mass properties of the CSM/LM for a specified configuration and can compute the CM trim aerodynamics for entry. Three programs are used to determine the RCS propellant status on either the CSM or LM, one is for determining the heat load generated by use of the extravehicular mobility unit, one monitors the super-critical helium pressure in the descent propulsion system, one is for diagnosis of the LM telemetry system, and the other monitors the LM electrical power system (EPS).

4.9.1 Mass properties and aerodynamics. - This program was developed to determine the CM entry aerodynamics in addition to the CSM/LM center of gravity location, moments of inertia and engine trim angles for any combination of the CSM and LM. The program has four options with which the CM entry aerodynamics, CM, CSM, LM, or CSM/LM center of gravity location, mass properties, and digital autopilot command loads can be generated.

4.9.1.1 Aerodynamics: This option computes the trim aerodynamic coefficients of the command module as a function of Mach number.

Inputs required:

a. Weight of present CM configuration

b. X, Y, and Z components of the center of gravity

Outputs required:

The outputs required are displayed on the aerodynamics update summary sheet shown in Figure 30.

4.9.1.2 Center of gravity: In addition to generating the CM entry trim aerodynamics, this option is used to compute the new center-of-gravity location for any desired configuration of the CSM/LM. Any of the five following configurations may be specified. With the SPS thrusting there are three options: SPS fuel at the bottom of the tank (undocked), SPS fuel at the bottom of the tank and both DPS and APS fuel at the top of the tank (CSM docked with the unstaged LM), and SPS fuel at the bottom and APS fuel at the top (CSM docked with the staged LM). With the DPS thrusting there are two options: DPS and APS fuel at the bottom of the tank (LM alone) or DPS/APS fuel at the bottom and SPS fuel at the tope of its tanks (CSM docked with the unstaged LM).

Inputs required:

a. Weight, center-of-gravity position, and moments of inertia of the CSM, LM, and modules to be added or subtracted

b. Total number of modules to be considered

Outputs required:

The outputs required are displayed on the center-of-gravity summary sheet shown in Figure 31.

4.9.1.3 Mass properties table: This option generates a table of center-of-gravity locations, moments of inertia, product moments of inertia, and engine trim angles as a function of the vehicle (docked or undocked) weight for a specific oxidizer-to-fuel mixture ratio. A portion of the table (c.g. position as a function of weight) is output on punched cards in a format acceptable to the RTCC and RTACF trajectory programs.

Inputs required:

a. Dry weight of the CM, SM, and LM ascent and descent stages

b. Weight of consumables

c. CM, SM, ascent stage and descent stage center-of-gravity locations, and moments of inertia

d. SPS, APS, and DPS oxidizer-to-fuel mixture ratio

e. Weight, center-of-gravity location, and moments of inertia of any items to be considered

Outputs required:

The outputs required are displayed on the mass properties summary sheet shown in Figure 32.

4.9.1.4 Digital autopilot command load: This option computes those mass properties required for uplink to the digital autopilot. The quantities are converted to the proper units and octal format acceptable to the CMC or LGC digital autopilot programs.

Inputs required:

a. Dry weight of CM, SM, and LM ascent and descent stages

b. Weight of consumables

c. CM, SM, ascent stage, and descent stage center-of-gravity locations and moments of inertia

d. SPS, APS, and DPS thrust levels

e. Weight, center-of-gravity location, and moments of inertia of any items to be considered

Outputs required:

The outputs required are displayed on the digital autopilot command load summary sheet shown in Figure 33.

4.9.2 Mass Properties, Reaction Control System, Service Propulsion System (MRS) Program. - The MRS Program is designed to generate a complete RCS propellant budget using premission data supplied for individual maneuver propellant consumption and internally computed mass properties characteristics. During the mission, as propellant is expended and vehicle configuration modified, the RCS portion of the program accepts inputs from the mass properties portion for use in its computations. In addition to mass properties, the RCS portion uses a form of flight timeline which is input by the user and fixed data which are stored in the program. The program is used in real-time mission support to correct the preflight budget in accordance with changes in the basic flight plan or procedure.

The input and output description of the MRS Program is given in the MRS Program description, which is presented in Reference 8.

4.9.3 Pressure, volume, temperature (PVT) equation for SM RCS. - This program determines the amount of SM RCS oxidizer and fuel remaining in each tank and how much of this can be considered useful propellant. From telemetered values of helium temperature and pressure, the program employs the gas equation to determine the volume of helium used to pressurize the fuel-oxidizer system. Once the volume of helium is determined in each tank, the amount of fuel or oxidizer is computed from the known total volume of each tank. The amount of usable propellant is then determined from the oxidizer-to-fuel mixture ratio being used and the expulsion efficiency of each tank.

Inputs required:

a. Volume of each oxidizer, fuel, and helium tank

b. Volume of all connecting lines

c. Initial pressure and temperature of the helium tank

d. Initial weight of fuel and oxidizer in each tank

e. Helium source pressure and temperature at the time of the PVT calculation

f. Fuel and oxidizer manifold pressure

g. Oxidizer-to-fuel mixture ratio

h. Weight of oxidizer and fuel remaining in the passive system (primary or auxiliary)

i. Expulsion efficiency of each tank

Outputs required:

The outputs required are displayed on the PVT summary sheet shown in Figure 34.

4.9.4 Extravehicular Mobility Unit (EMU) processor. - This processor will be used to determine the heat load on the EMU coolant water loop while one astronaut is performing extravehicular activity.

The processor uses four methods and each result is output on the summary sheet. Method 1, the metabolic rate method, uses an equation that relates the amount of oxygen used by the EVA astronaut to the amount of heat generated. Method 2 relates the mass flow rates of the ventilation loop, sublimator, and coolant water loop to determine heat generated. Method 3 determines the heat generated by direct read off from a plot relating heat load to the feed pressure of the water loop. The last method converts the heat loads from the other three methods from Btu/hour to pounds of water used in the loop per hour.

Inputs required:

a. PLSS primary oxygen subsystem decay rate

b. EMU leak rate

c. Total heat into the system from equipment operation

d. Total heat into the system from environment

e. Heat leak from suit environment

f. Heat from lungs, perspiration, respiration, etc.

g. Sublimator loop mass flow rate

h. LCG H_2O inlet temperature

i. LCG H_2O outlet temperature

j. Total H_2O transport loop mass flow rate

k. Efficiency number

l. Heat output from the reaction of CO_2 with LiOH

m. Feed water pressure

Outputs required:

The outputs required are displayed on the EMU processor summary sheet shown in Figure 35.

4.9.5 LM RCS propellant status. - This program is designed to generate a computed RCS propellant budget for the lunar module. It was developed from the MRS Program which serves the same functions for the CSM. The comments in Section 4.9.2 also apply to this section.

Inputs required:

The inputs to the LM RCS Program are given in Reference 9.

Outputs required:

The outputs of the program are displayed on the LM RCS propellant budget summary sheet shown in Figure 36.

4.9.6 DPS supercritical helium (SH_e) pressure profile. - This program will be used to monitor the LM DPS supercritical helium propellant tank pressurization system. This system provides pressure for the DPS engine fuel feed. As fuel is used by the engine, it is replaced by helium that is stored in a high-density gas in the supercritical state. The helium pressure is maintained by increasing the temperature in its storage bottle as the fuel is used.

Heat is transferred from the fuel to the helium bottle by means of heat exchangers. This program accepts the mission profile, consisting of various burns and coast phases, and computes the corresponding helium pressure resulting from heat transfer between the systems.

Inputs required:

The inputs to this program are described in Reference 10.

Outputs required:

The outputs of the program are displayed on the SH_e summary sheet shown in Figure 37.

4.9.7 LM Telemetry Diagnostics Program. - This program will accept a list of onboard sensors that have detected failures on the LM and output a summary sheet report that designates the common failure mode of the list of input parameters.

Inputs required:

List of sensors

Outputs required:

The outputs required are displayed on the LM telemetry processor summary sheet shown in Figure 38.

4.9.8 Spacecraft Electrical Energy Network Analysis (SEENA) Program. - This program will be used to determine the capability of the LM EPS to support the various phases of the mission. It determines: (1) the total energy drain on the LM power supply for any configuration of the onboard electrical equipment, and (2) the remaining energy available based on an event timeline. This timeline of chronological events defines changes in the EPS switching status which in turn defines the total load on the system. The premission data (power distribution network, battery characteristics, and power requirements of components) are stored internally, and as these components are added to or removed from the circuit by on-line card input to the program, new total energy drain and the resultant energy available are computed by the program and are output as printed data of consumables (energy consumed, battery charge status) and steady state conditions (components ON, voltages, currents, etc.) of the EPS circuit. In addition, tapes may be written (optional) for data plotting and for thermal data to be input to an environmental control system (ECS) computer program.

The input and output description of the SEENA Program is detailed in the SEENA user's guide which is presented in Reference 11.

4.10 Deorbit Processors

There are eight deorbit processors in the RTACF to support the Apollo 9 mission. Two processors were developed to simulate deorbits to primary landing areas while deorbits to contingency landing areas required the use of five processors, two of which simulate hybrid deorbits. The eighth processor is a block data program designed to generate a series of deorbits including CLA's, PLA's, and SM RCS apogee deorbits.

4.10.1 Primary landing area. - The PLA processors were developed to simulate a targeted deorbit into a primary landing area. This area is a region about a predetermined landing point where recovery forces have been stationed or can be made available in a short period of time. The target point in the processors is defined by a latitude and longitude, and the processor iterates on both the deorbit ignition time and the time to reverse the bank angle to achieve the specified target.

4.10.1.1 PLA processor 1: This processor employs an iterative technique to determine the SPS or RCS deorbit ignition time and time to reverse bank to land in a primary landing area. The deorbit maneuver is determined by the following quantities: an initial attitude, a guidance mode, and a terminating value for the maneuver. The initial vehicle attitude can be input with respect to the local vertical/local horizontal (LV/LH) system or aft-looking line of sight to the horizon. The attitude may also be specified by indicating the initial thrust vector attitude with respect to the LV/LH system or by indicating the IMU gimbal angles and stable member alignment. The spacecraft orientation, during the maneuver, is maintained inertial by either the stabilization and control system (SCS) or external ΔV guidance mode. The maneuver thrust may be terminated after a specified value of one of the following conditions has been satisfied: an incremental velocity change, external ΔV onboard targets, a burn duration, or a velocity and flight-path angle constraint at entry interface.

There are two entry profiles that can be used to achieve a target latitude and longitude. One profile maintains a constant bank angle from entry interface to the computed time to reverse bank, followed by the negative of the initial bank angle to drogue chute deployment. The other profile employs the following sequence: a constant lift vector orientation to a specified g-level, a constant bank angle to the time to reverse bank, and the negative of the previous bank angle to drogue chute deployment.

Inputs required:

a. RTCC state vector

b. CSM weight

c. CM weight at entry

d. REFSMMAT or IMU gimbal angles*

e. Deorbit SPS or RCS maneuver data

f. Ignition time (initial guess)

g. Entry profile

h. Latitude and longitude of target

---

*If REFSMMAT is input, the IMU gimbal angles will be output. If IMU gimbal angles are input, the required REFSMMAT will be output.

Outputs required:

a. Deorbit ignition time

b. Total ΔV along the X-body axis less tailoff

c. External ΔV components at ignition

d. Duration of the burn

e. IMU gimbal angles at ignition (if not input)

f. Latitude, longitude, altitude, and true anomaly at ignition

g. Thrust vector pitch at ignition

h. Velocity, flight-path angle, and g.e.t. at 400,000 feet

i. Time from retrofire to 400,000 feet

j. Time from retrofire to a specified number of g's

k. Blackout data

l. Maximum g-level

m. Time from retrofire to drogue chute deployment

n. Time from retrofire to main chute deployment

o. Latitude, longitude, and g.e.t. of landing

p. Time from retrofire to the time of reverse bank

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.1.2 PLA processor 2: This processor is essentially the same as the PLA processor 1 with the additional capability of simulating the S-IVB/CSM separation maneuver. The separation maneuver can be simulated at either a specified time or at a fixed time prior to the deorbit maneuver. The fixed-time separation, using the SPS or RCS thrusters, is computed based on the desired time of ignition, the spacecraft or thrust vector attitude, incremental velocity to be added, and either the manual thrust vector control (MTVC) or SCS guidance mode. Simulating the separation at a fixed-time interval before the deorbit maneuver requires the same inputs as the fixed-time separation except for the ignition time. Instead of specifying the time of ignition, a constant Δt between the separation and deorbit maneuvers is input. The processor then employs an iterative technique to determine the ignition time of the separation maneuver and, hence, the deorbit ignition time.

Inputs required:

The inputs (a-h) required are identical to PLA processor 1 with the following additions:

i. S-IVB/CSM weight before separation

j. SPS or RCS separation maneuver data

Outputs required:

The outputs (a-p) required are identical to PLA processor 1 with the following additions:

q. Separation maneuver time

r. Time from separation to retrofire

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.2 Contingency landing area. - In the event of an inability to deorbit into a primary landing area, any suitable area along the groundtrack can be designated as a contingency landing area. The CLA processors can simulate a deorbit to a contingency landing area by targeting only for a longitude. The deorbit ignition time is varied to achieve the target longitude using a fixed entry profile.

4.10.2.1 CLA processor 1: This CLA processor employs an iterative technique to determine the SPS or RCS deorbit ignition time to hit a target longitude in a contingency landing area using a fixed entry profile. The deorbit maneuver, like PLA processor 1 described above, is determined by an initial attitude, a guidance mode, and a terminating value for the maneuver. The entry profile used may be any one of the following:

a. A constant bank angle maintained from entry interface to drogue chute deployment

b. A lift vector orientation from entry interface to a specified g-level, followed by a constant bank angle to drogue chute deployment

c. A rolling entry profile either maintaining a constant bank angle from entry interface to 300,000 feet or a constant lift vector orientation from entry interface to a specified g-level, followed by a rolling entry to 75,000 feet and then a full lift attitude to drogue chute deployment

Inputs required:

The inputs (a-g) required are identical to the PLA processor 1 with the following exception:

h. Longitude of target

Outputs required:

The outputs required are identical to the PLA processor 1 except that the time from retrofire to the time to reverse bank angle does not apply.

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.2.2 CLA processor 2: This processor is essentially the same as the CLA processor 1 described above, with the additional capability of simulating the S-IVB/CSM separation. The separation can occur at either a fixed time or fixed Δt before the maneuver as described in the PLA processor 2.

Inputs required:

The inputs (a-g) required are identical to the PLA processor 1 with the following additions:

h. Longitude of target

i. S-IVB/CSM weight before separation

j. SPS or RCS separation maneuver data

Outputs required:

The outputs (a-o) required are identical to the PLA processor 1 with the following additions:

p. Separation maneuver time

q. Time from separation to retrofire

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.2.3 CLA processor 3: This processor employs an iterative technique to determine the CM RCS ignition time required to deorbit the CM to a CLA. The maneuver can be performed in either the SCS or MTVC guidance mode with the ignition time determined by the incremental velocity to be added, initial pitch attitude, and the target longitude. The CM entry profile will be defined by the following sequence: a constant lift vector orientation from entry interface to a specified g-level will be flown, followed by a constant roll rate to an altitude of 75,000 feet, and then full lift to drogue chute deployment.

Inputs required:

Same as for CLA processor 1

Outputs required:

Same as for CLA processor 1

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.3 Hybrid deorbit. - In the event of an inability to use the SPS engine, the hybrid deorbit processors simulate a deorbit using: (1) the SM and CM RCS thrusters or (2) the LOX dump maneuver and the CM RCS thrusters. The engine may be inoperative because of an inability to start it or because of a failure to separate the CSM from the S-IVB. Two processors have been developed to simulate these conditions.

4.10.3.1 Hybrid deorbit processor 1: If a situation arises where the SPS engine cannot be used to deorbit the CSM and there is insufficient SM RCS propellant to perform the RCS deorbit, a hybrid deorbit will be performed. The hybrid deorbit consists of an SM RCS burn, CM/SM separation, and a CM RCS burn to achieve a deorbit to a specified CLA.

This hybrid deorbit processor employs an iterative scheme to determine the RCS ignition times in order to achieve a target longitude. The SM RCS burn duration is based on a fixed incremental velocity to be realized by the SM RCS burn. The spacecraft maintains an inertial attitude which is specified in the LV/LH plane at the centroid of the hybrid deorbit. A 60-second coast between the SM and CM RCS burns allows time to perform the CM/SM separation and the reorientation of the CM for the CM RCS maneuver. The CM attitude is maintained so that the thrust vector alignment is the same as the SM RCS thrust vector alignment. During entry, a specified lift vector orientation is maintained to a fixed g-level, followed by a specified lift entry profile to drogue chute deployment.

This processor can also be used to evaluate the actual hybrid deorbit performed. The SM RCS maneuver is assumed to have performed nominally while the CM RCS maneuver is defined by the actual incremental velocities achieved. The processor remains essentially unchanged except that the SM RCS ignition time and actual velocity increments are input and the landing point is determined.

Inputs required:

Same as CLA processor 2

Outputs required:

Same as CLA processor 2

Optional outputs:

Footprint data

The outputs required for this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.3.2 Hybrid deorbit processor 2: A hybrid deorbit will be performed in the event of a failure to deploy the SLA panels and a resultant inability to separate the CSM from the S-IVB. The deorbit will consist of dumping the onboard LOX to obtain a retrograde delta velocity, the CM separation from the SM/S-IVB, and a CM RCS burn at the following apogee to achieve a landing in a CLA.

This hybrid deorbit processor uses an iterative technique to determine the CM delta velocity required to achieve the target longitude, given the ignition time and delta velocity of the LOX dump maneuver and the time of the CM burn initiation (apogee). The LOX dump maneuver will be performed in the constant orbital rate mode which maintains a constant pitch attitude to the local horizontal. After the LOX dump maneuver, the CM separates from the SM/S-IVB and assumes a thrust vector attitude which is specified in the LV/LH plane at the centroid of the hybrid deorbit. The CM RCS maneuver is performed by maintaining the thrust vector attitude inertial. The entry profile consists of a lift vector orientation to a specified g-level followed by a specified lift vector orientation to drogue chute deployment.

Inputs required:

a. RTCC state vector

b. S-IVB/CSM weight

c. CM weight at entry

d. REFSMMAT or IMU gimbal angles*

e. Time of LOX dump initiation

f. Entry profile

g. Longitude of target

h. Delta velocity of LOX dump maneuver

i. Apogee at which CM burn is to be performed

Outputs required:

The outputs (a-o) are the same as PLA processor 1 with the following additions:

p. Time of CM RCS ignition

q. Delta velocity of CM RCS maneuver

The outputs of this processor will be displayed on the standard summary sheet shown in Figure 39.

4.10.4 Block data. - Block data nominally consist of a series of deorbit maneuvers which can be executed at discrete times during a period of six revolutions to deorbit the CSM in the event of a contingency requiring rapid mission termination. The data are generated and sent to the crew in blocks of six revolutions with three types of deorbits per revolution: an SPS deorbit to a PLA, an SPS deorbit to a CLA, and a SM RCS apogee deorbit.

The Apollo Block Data Program (ABDP) has the capability of performing any number of deorbit maneuvers with or without orbital maneuvers. This program uses an iterative scheme to determine the deorbit ignition time to achieve the target landing point. An orbital maneuver can be performed by specifying the ignition time, the spacecraft attitude, the guidance mode to be used, and an incremental velocity

*If REFSMMAT is input, the IMU gimbal angles will be output. If IMU gimbal angles are input, the required REFSMMAT will be output.

or external ΔV components. The deorbit maneuver is nominally specified by a velocity and flight-path angle constraint at entry interface, spacecraft attitude with respect to the aft-looking line of sight to the horizon, the SCS guidance mode, and a PLA or CLA entry profile. The inputs required to execute the ABDP and the output data and summary sheets, which can be generated, are described in the "Apollo Block Data Program User's Manual" (Reference 12).

4.11 Guided Entry and Backup Guidance Quantities Processors

Guided entries are those whose steering commands are controlled by the entry logic in the onboard computer. As a backup capability to the guidance system, the commander monitors the entry using the entry monitor system display and the flight director attitude indicator. He may take control of the entry at any point and manually fly the CM to touchdown using ground computed backup guidance quantities.

The Apollo Reentry Simulation (ARS) Program takes a state vector at 425, 000 feet and computes the necessary entry profile to achieve a specified target latitude and longitude. The state vector is generated by one of the ARMACR deorbit maneuver processors which can either simulate the deorbit maneuver, given a preburn state vector, or propagate a postburn state vector to 425, 000 feet. At this point, the necessary data are saved and then used by the ARS Program to compute the guided entry or necessary backup guidance quantities. There are six entry steering modes in the ARS Program. A description of these steering modes is presented below.

4.11.1 Automatic guidance and navigation control. - In this steering mode, the ARS Program uses the CMC entry logic to compute the entry steering commands and to simulate the entry trajectory required to achieve the target landing point.

4.11.2 Open loop followed by guidance and navigation control. - In this entry mode, an initial bank angle is maintained from 400, 000 feet to a specified g-level, at which time the CM is rolled to a second bank angle designated as the backup bank angle. This attitude is maintained until a second g-level is reached. From this time until drogue chute deployment, the ARS Program uses the guidance and navigation control logic to compute the steering commands necessary to achieve the target landing point. This steering mode requires the input of an initial and backup bank angle and two g-levels.

4.11.3 Bank-reverse-bank. - In this entry mode, which is used to compute backup guidance quantities, an initial bank angle is maintained from 400, 000 feet to a specified g-level. It is then followed by a backup bank angle to a computed time to reverse bank, and then the reverse bank angle is flown to drogue chute deployment. In this steering mode, the initial bank angle and g-level are input, and the backup bank angle and time to reverse bank are computed by the ARS Program.

4.11.4 Combined bank-reverse-bank and guidance and navigation control. - This entry mode is the same as that described in the second steering mode with the exception that the program computes the backup bank angle. The inputs consist of the initial bank angle and the two g-levels.

4.11.5 Rolling. - In this entry, an initial bank angle is maintained from 400, 000 feet to a specified g-level, followed by a constant roll rate to drogue chute deployment. This mode requires the input of the initial bank angle, g-level, and roll rate.

4.11.6 Open loop. - This entry can either be a bank-reverse-bank, as described in the third steering mode, or a constant bank angle entry from 400, 000 feet to drogue chute deployment. The bank-reverse-bank option of this steering mode requires the input of the initial and backup bank angles, the g-level, and the time to reverse bank. A constant bank angle entry can be specified by inputting the value of the bank angle to be used as the initial bank angle and inputting the g-level and time to reverse bank as large values.

Inputs required:

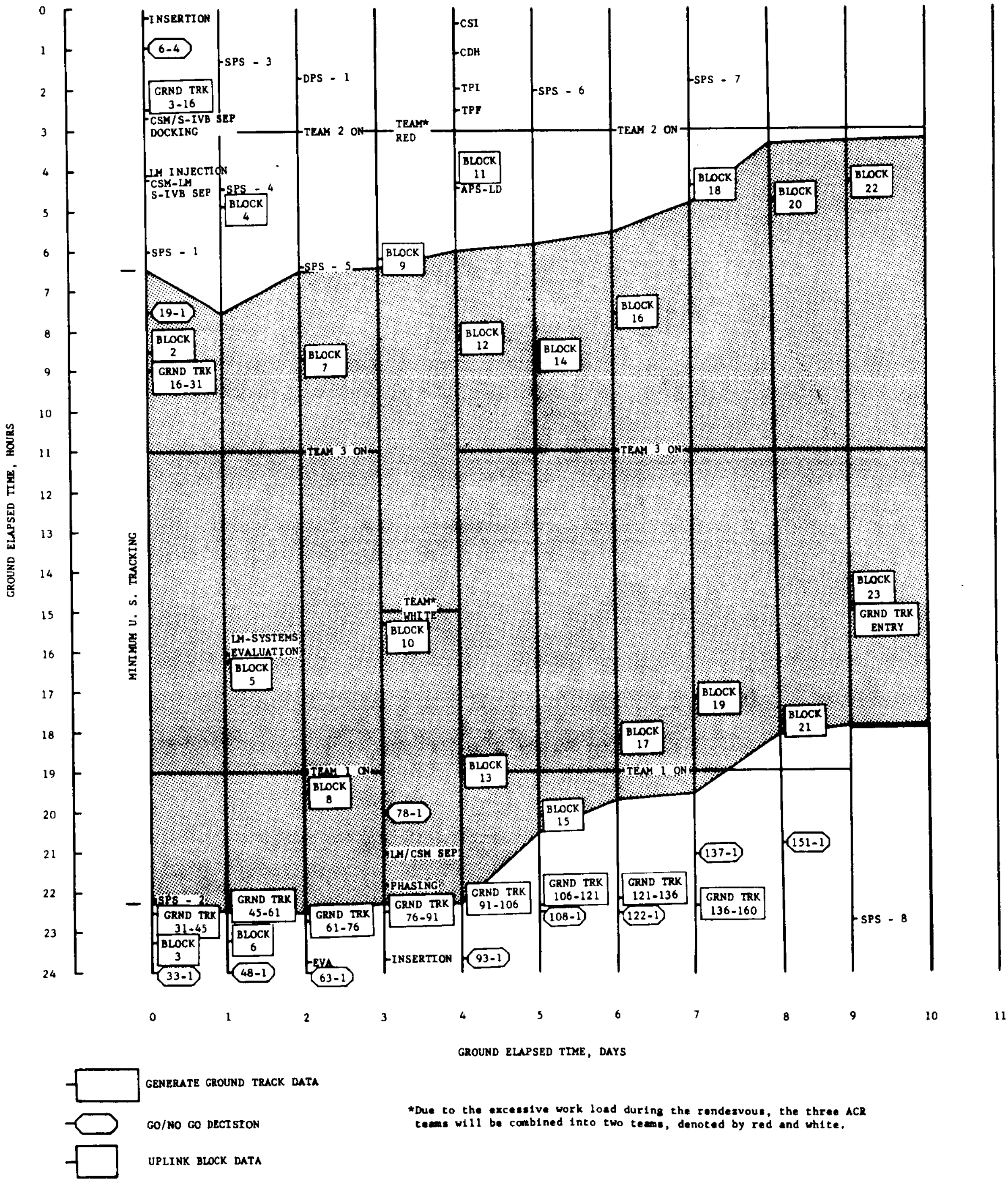
The steering mode and necessary entry quantities

Outputs required:

The outputs required for this processor are displayed on the ARS summary sheet shown in Figure 40.

5. APOLLO 9 RTACF NOMINAL MISSION TIMELINE

This section of the Flight Annex presents the RTACF nominal mission timeline. The timeline is not meant to display the schedule of RTACF activities during the mission but rather to present, in graphical form, a synopsis of the events that are of particular importance in planning the RTACF work schedule, which is to be published as a separate MSC internal note. The events and their corresponding times presented in the timeline include the major nominal mission events, the region where U.S. tracking capability exists, the go/no-go decisions, block data uplink, and the generation of new ground track data. The event times used in generating the timeline were based on the Apollo 9 Operational Trajectory (Revision 1).



*Due to the excessive work load during the rendezvous, the three ACR teams will be combined into two teams, denoted by red and white.

Apollo 9 RTACF Nominal Mission Timeline

6. APOLLO 9 RTACF OPERATIONAL SUPPORT TEAM

The following individuals will man positions in the Flight Dynamics Staff Support Room and the Auxiliary Computing Room during the Apollo 9 mission.

Flight Dynamics Staff Support Room

	1st Team	2nd Team	3rd Team
Trajectory Support Chief	S. D. Holzaepfel (FAB)	T. L. Turner (FAB)	H. Garcia (FAB)
Assistant Trajectory Support Chief	R. D. Davis (FAB)	L. D. Davis (FAB)	E. R. Hischke (FAB)
Maneuver Specialist	H. L. Conway (OMAB)		R. S. Merriam (OMAB)
Rendezvous Specialist	J. D. Alexander (OMAB)		L. D. Hartley (OMAB)
Entry Specialist	J. C. Harpold (LAB)		O. Hill (LAB)
Data Management	M. A. Collins, Jr. (MPSO)		R. E. Simms (MPSO)
Engineering Aide	S. L. Hallmark (ITT)	J. L. Snyder (ITT)	C. A. Henson (ITT)

Auxiliary Computing Room

ACR Chief	P. A. DiValerio (TRW)	H. L. Sanders (TRW)	V. R. Dragotta (TRW)
ACR Engineers	D. C. McDougall (TRW) D. W. Sager (TRW) B. G. Schneider (FAB)	L. Baker, Jr. (TRW) R. T. Witton (TRW) T. W. Locke (TRW)	J. H. Kawasaki (TRW) C. D. Chenoweth (TRW) R. Moore (TRW)
Program Consultants	K. Christie (TRW) R. Stein (LEC)	R. Corley (LEC)	R. Baker (LEC)
Run Coordinators	J. E. Friebele (LEC) R. Gregory (LEC)	B. N. Ferguson (LEC) L. Burney (LEC)	W. Baker (LEC) D. Lacefield (LEC)
Engineering Aide	M. D. Johnson (ITT)	S. L. Park (ITT)	P. J. Nix (ITT)

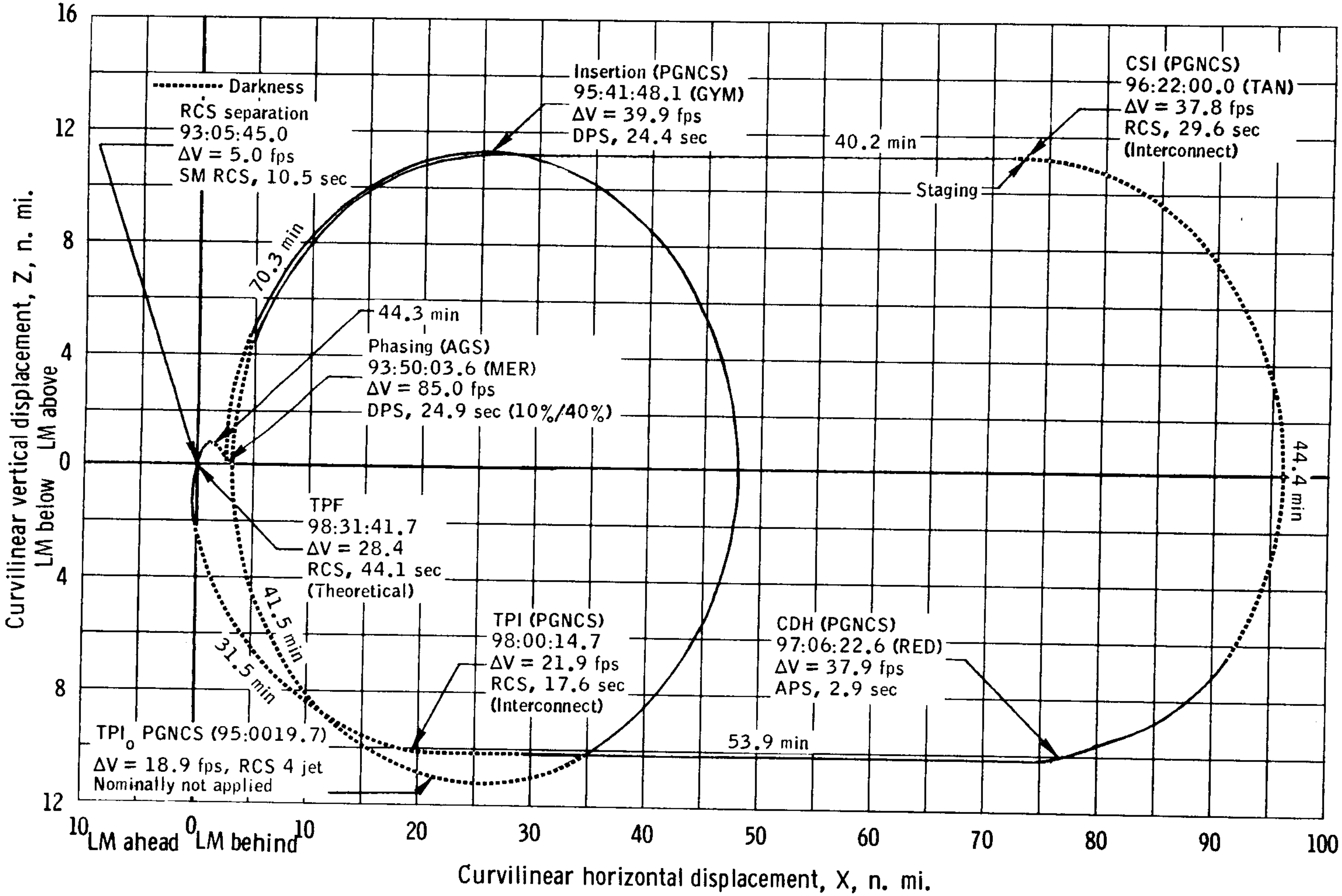


Figure 1. Rendezvous Sequence Relative Print Plot

MODE I LAUNCH ABORT DATA

TA SEC	RANGE FT	RANGE NM	BEAR[1] E/N	RAN E FT	RAN N FT	GET IP SEC	LAT-D DEG	LONG DEG	MAX ALT FT-MSL	DD ALT FT	MD ALT FT
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX
XX	XXXXXX	XXXX	XXX	XXXXX	XXXXX	XXX	XXXX	XXXX	XXXXX	XXXXX	XXXXX

[1] MINUS SIGN INDICATES WEST OF NORTH (W/N)

Figure 2. Mode I Launch Abort Summary Sheet

FDO ORBIT DIGITALS

GET	XXX:XX:XX		LON PP	XXX:XX	E/W
REV	XXX		LAT PP	XX:XX	N/S
			GET CC	XXX:XX:XX	
STA ID			T-A	XXXX	
			LON AN	XXX:XX	E/W
H	XXXX				
VI	XXXX				
GAM I	XXXX				
A	XXXX				
E	XXXX				
I	XXXX				
HA	XXXX				
LAT A	XX:XX	N/S			
LON A	XXXX:XX	E/W			
GET A	XXXX:XX:XX				
HP	XXXX				
LAT P	XX:XX	N/S			
LON P	XXXX:XX	E/W			
GET P	XXXX:XX:XX				

Figure 3. FDO Orbit Digitals Summary Sheet

RELATIVE MOTION DIGITALS

CSM STA ID VEH LM STA ID

REFSMMAT X AXIS AT TARGET

GET	R	RDOT	AZH	ELH	X/P	Z/Y	Y/R
XXX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
					XXX	XXX	XXX
XXX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
					XXX	XXX	XXX
.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.

Figure 4. Relative Motion Summary Sheet

RADIATION EVALUATION

LONG DEG	LAT DEG	ALT NM	B GAUSS	L ER	DAY	HR	MIN	SEC	DOSE RATE (REM/HR) LM SKIN	CM SKIN	CM DEPTH	CUMULATIVE DOSE (REM) LM SKIN	CM SKIN	CM DEPTH
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
XXXX	XXXX	XXXX	XXXX	XXXX	XX	XX	XX	XX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
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.	.	.	.	.	.	.	.	.	.	.	.	.	.	.

Figure 5. Radiation Evaluation Summary Sheet

GROUND TRACK

REV NO.	GET HRS	GET MIN	GET SEC	GMT DAY	GMT HRS	GMT MIN	GMT SEC	LAT DEG	LAT MIN	LONG DEG	LONG MIN	ALT N MI	AZIMUTH DEGREES
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXXX	XXXXX
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Figure 6. Ground track Summary Sheet

RTACF

DOCKED CSM GIMBAL ANGLES COMPUTATION

XXX GIMBAL ANGLES ARE COMPUTED

	CSM IMU GIMBAL ANGLES			LM IMU GIMBAL ANGLES			
REFSMMAT ID	IGA(P) DEG	MGA(Y) DEG	OGA(R) DEG	IGA(P) DEG	MGA(R) DEG	OGA(Y) DEG	REFSMMAT ID
LIFTOF XXXXX	XXX	XXX	XXX	XXX	XXX	XXX	LIFTOF XXXXX

Figure 7. LM/CSM Docking Alignment Summary Sheet

EVA OPEN HATCH THERMAL CONTROL ALIGNMENT

GROUND ELAPSED TIME			IMU GIMBAL ANGLES			SUN LOOK ANGLES	
HRS	MIN	SEC	IGA(P) DEG	MGA(Y) DEG	OGA(R) DEG	THETA DEG	PHI DEG
XXX	XX	XX	XXX	XXX	XXX	XXX	XXX

REFSMMAT ID = XXXXXXXXXXXX

XIMUX = XXXXXXXX	XIMUY = XXXXXXXX	XIMUZ = XXXXXXXX
YIMUX = XXXXXXXX	YIMUY = XXXXXXXX	YIZUM = XXXXXXXX
ZIMUX = XXXXXXXX	ZIMUY = XXXXXXXX	ZIMUZ = XXXXXXXX

Figure 8. EVA Open Hatch Summary Sheet

FDO DETAILED MANEUVER TABLE

```
C STA ID                      L STA ID                    STA ID                  WT      XXXXX
GMTV    XX XX XX XX           GMTV   XX XX XX XX          GMTV  XX XX XX XX       WC      XXXXX
GETV    XXX  XX  XX           GETV   XXX  XX  XX          GETV  XXX  XX  XX       WL      XXXXX
CODE                          REF                         GETR  XXX  XX  XX       WF      XXXXX

GETI    XXX  XX  XX           DT B           XXX   DT TO    XXX    REFSMMAT       DEL P   XXX
PETI    XXX  XX  XX           DT U           XXX   DV TO    XXX                   DEL Y   XXX
DVM        XXXX            VGX     XXX            OR     XXX      YB   XXX       YH      XXX
DVREM      XXXX            VGY     XXX            IP     XXX      PB   XXX       PH      XXX
DVC        XXXX            VGZ     XXX            MY     XXX      RB   XXX       RH      XXX

VF         XXXX            H  BI   XXX            HA     XXX      VP             XXX
VS         XXXX            P  BI   XXX            HP     XXX      THETA P        XXX
VD         XXXX            L  BI   XXX            L  AN  XXX      DELTA P        XXX
DH          XXX            F  BI   XXX            E      XXX      P LLS          XXX
PHASE        XX                                   I     XXXX      L LLS          XXX
PHASE DOT    XX                                   WP    XXXX      R LLS          XXX
WEDGE ANG   XXX                  UNTIL
YD          XXX

                                       TARGETS

PGNS                        AGS
                                                          LAMBERT
EXT DV CSM                  EXT    DV                                         MVR     XXX  XX  XX
GETI    XXX  XX  XX         GETI        XXX  XX  XX  GETI   XXX  XX  XX      GETI       XXX
   VX       XXX                VX           XXX        T F  XXX  XX  XX      APSIS      XXX
   VY       XXX                VY           XXX          X       XXX         ELEV    XXX  XX  XX
   VZ       XXX                VZ           XXX          Y       XXX         TPI     XXX  XX  XX
                                                         Z       XXX         DT
                                                         C       XXX         OPTION

WT AFTER XXXXX
```

Figure 9. FDO Detailed Maneuver Table

MANEUVER EVALUATION OUTPUT

DELTA V	TOTAL=	XXXXX	
	DELTA VX=	XXXXX	
	DELTA VY=	XXXXX	
	DELTA VZ=	XXXXX	

PITCH= XXXXX YAW= XXXXX

GIMBAL ANGLES

OUTER=	XXXXX
INNER=	XXXXX
MIDDLE=	XXXXX

Figure 10. Maneuver Evaluation Summary Sheet

NAV UPDATE TO CMC OR LGC

LOAD NO GET GEN SITES

STA ID GMT ID

OID	FCT	DSKY V71	VECTOR
1	INDEX		
2	ADD		
3	VID		
4	X	XXXXX	XXXXXXX
5	X	XXXXX	
6	Y	XXXXX	XXXXXXX
7	Y	XXXXX	
10	Z	XXXXX	XXXXXXX
11	Z	XXXXX	
12	X-DOT	XXXXX	XXXXXXX
13	X-DOT	XXXXX	
14	Y-DOT	XXXXX	XXXXXXX
15	Y-DOT	XXXXX	
16	Z-DOT	XXXXX	XXXXXXX
17	Z-DOT	XXXXX	
20	T	XXXXX	XXX XX XX
21	T	XXXXX	

Figure 11. Command Load Navigation Update Summary Sheet

CMC (OR LGC) EXT DELTA-V UDPATE

LOAD NO SITES

STA ID GMT ID

GET GEN MAN CODE

OID	FCT	DSKY V71	DECIMAL
1	INDEX		
2	ADD		
3	TIGN	XXXXX	XXX XX XX
4	TIGN	XXXXX	
5	VGX	XXXXX	XXXXX
6	VGX	XXXXX	
7	VGY	XXXXX	XXXXX
10	VGY	XXXXX	
11	VGZ	XXXXX	XXXXX
12	VGZ	XXXXX	
13	MAN WT	XXXXX	XXXXX

Figure 12. Orbital External ΔV Summary Sheet

CMC RETROFIRE EXTERNAL DELTA-V UPDATE

TTI	STA ID		TYPE	LOAD NO		
CT	GET GEN		SITES PRI		B/U	
THRUSTER		OID	FCT	DSKY V71	DECIMAL	
AREA		1	INDEX			
		2	ADD			
GMTI		3	LAT	XXXXX	XXXXX	DEG
GETI		4	LAT	XXXXX		
		5	LONG	XXXXX	XXXXX	DEG
BT		6	LONG	XXXXX		
V(C)		7	TIGN	XXXXX	XXX XX XX	
		10	TIGN	XXXXX		
R(O)		11	VGX	XXXXX	XXXXX	FPS
P(I)		12	VGX	XXXXX		
Y(M)		13	VGY	XXXXX	XXXXX	FPS
		14	VGY	XXXXX		
H(P)		15	VGZ	XXXXX	XXXXX	FPS
RT400K		16	VGZ	XXXXX		
RETRB		17	WT	XXXXX	XXXXX	LBS

Figure 13. Deorbit External ΔV Summary Sheet

REFSMMAT UPDATE FORMAT

MAT OID	FCT	DSKY	DECIMAL
01	INDEX		
02	ADD		
03	XIXE	XXXXX	XXXXXXX
04	XIXE	XXXXX	
05	XIYE	XXXXX	XXXXXXX
06	XIYE	XXXXX	
07	XIZE	XXXXX	XXXXXXX
10	XIZE	XXXXX	
11	YIXE	XXXXX	XXXXXXX
12	YIXE	XXXXX	
13	YIYE	XXXXX	XXXXXXX
14	YIYE	XXXXX	
15	YIZE	XXXXX	XXXXXXX
16	YIZE	XXXXX	
17	ZIXE	XXXXX	XXXXXXX
20	ZIXE	XXXXX	
21	ZIYE	XXXXX	XXXXXXX
22	ZIYE	XXXXX	
23	ZIZE	XXXXX	XXXXXXX
24	ZIZE	XXXXX	

Figure 14. REFSMMAT Update Summary Sheet

ENGINEERING UNITS/OCTAL CONVERSION

ENGINEERING NUMBER	OCTAL NUMBER	POWER OF TWO	PRECISION	SCALE FACTOR
XXXXXX	XXXXX XXXXX	XX	XX	XXXXX

Figure 15. General Octal Conversion Summary Sheet

SATURN COMMAND LOAD

NAVIGATION UPDATE

LOAD NO XXXX

GETSV XX XX XX

FCT	ENGLISH	MATRIX	OCTAL LOAD DATA				
Z DOT	XXXXX	XXXXX	11 XXX	12 XXX	13 XXX	14 XXX	15 XXX
X DOT	XXXXX	XXXXX	21 XXX	22 XXX	23 XXX	24 XXX	25 XXX
Y DOT	XXXXX	XXXXX	31 XXX	32 XXX	33 XXX	34 XXX	35 XXX
Z POS	XXXXX	XXXXX	41 XXX	42 XXX	43 XXX	44 XXX	45 XXX
X POS	XXXXX	XXXXX	51 XXX	52 XXX	53 XXX	54 XXX	55 XXX
Y POS	XXXXX	XXXXX	61 XXX	62 XXX	63 XXX	64 XXX	65 XXX
TIME	XX XX XX	XXXXX	71 XXX	72 XXX	73 XXX	74 XXX	75 XXX

Figure 16. S-IVB Navigation Vector Update Summary Sheet

RTACF LM NAV UPDATE TO AGS

VECTOR ID	
VECTOR GET	XXX XX XX
VECTOR GMT	XX XX XX XX

X X	XXXXXXX
Y Y	XXXXXXX
Z Z	XXXXXXX
X-DOT X-DOT	XXXXXXX
Y-DOT Y-DOT	XXXXXXX
Z-DOT Z-DOT	XXXXXXX
T T	XXX XX XX
GETK GETK	XXX XX XX

Figure 17. AGS Navigation Update Summary Sheet

GETAC	XXX:XX:XX	PITCH	XXXXX	YAW	XXXXX	ROLL	XXXXX

	GIVEN VALUES		SMOOTHED VALUES	
SEXTANT	SHAFT	TRUNNION	SHAFT	TRUNNION
S1 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
S2 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

SCANNING TELESCOPE

STAR				M	R	LAT	LONG
1	XX	SHAFT	XXX	XX	XX	XXX	XXX
2	XX	TRUNNION	XXX	XX	XX	XXX	XXX

BORESIGHT

STAR	RHO	THETA	LAT	LONG
S1 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
S2 XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

MATRIX

XIXE	XXXXX	XIYE	XXXXX	XIZE	XXXXX
YIXE	XXXXX	YIYE	XXXXX	YIZE	XXXXX
ZIXE	XXXXX	ZIYE	XXXXX	ZIZE	XXXXX

Figure 18. Standard OST Summary Sheet

STAR SIGHTING TABLE

TGT ID		R-O	XXX		
TGT DEC	XX:XX:XX	P-I	XXX	LOS DEC	XX:XX:XX
TGT RT ASC	XXX:XX:XX	Y-M	XXX	LOS RT ASC	XXX:XX:XX
GND. PT DATA		SFT	XXX	REV	XXX
		TRN	XXX	LAT-LOS	XXXXX N/S
LAT	XXXXX N/S			GET-LOS	XXX:XX:XX
LONG	XXXXX E/W				
ALT	XXXXX			W-D	XXXXX
ELV	XXXXX			GETCA	XXX:XX:XX

REFSMMAT

XIX	XXXXXX	YIX	XXXXXX	ZIX	XXXXXX
XIY	XXXXXX	YIY	XXXXXX	ZIY	XXXXXX
YIZ	XXXXXX	YIZ	XXXXXX	ZIZ	XXXXXX
THETA-X	XXXXX	THETA-Y	XXXXX	THETA-Z	XXXXX

Figure 19. Star Sighting Table

STEERABLE ANTENNA POINTING PROGRAM

GET HR:MIN:SEC	P/T DEG	Y/S DEG	IGA DEG	MGA DEG	OGA DEG	SLANT RANGE N MI	AZMUTH	ELEV
XX XX XX	XXX	XXX	XXX	XXX	XXX	XXXXX	XXX	XX
XX XX XX	XXX	XXX	XXX	XXX	XXX	XXXXX	XXX	XX
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Figure 20. Steerable Antenna Pointing Summary Sheet

RADAR TRACKING

VEHICLE 1

STATION	CODE	REV NO.	ACQUISITION GET HR	MIN	SEC	ACQUISITION GMT DAY	HR	MIN	SEC	LOSS OF SIGNAL GET HR	MIN	SEC	LOSS OF SIGNAL GMT DAY	HR	MIN	SEC	DELTA T HR	MIN	SEC	MAX ELEV DEGREES	ACQ AZIMUTH DEGREES	ACQ RANGE N.MI.	MIN RANGE N.MI.
HTV	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
CAL	R V	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GDS	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GYM	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
WHS	C	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
TEX	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
PAT	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
MLA	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
GBI	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
ANT	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
BDA	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
VAN	R	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
CYI	RTVC	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
ASC	R V	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XXXX	XXXX
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Figure 21. Radar Summary Sheet

SPACECRAFT DAYLIGHT-DARKNESS

VEHICLE 1

	GET			GMT				GEODETIC LATITUDE		LONGITUDE		ALTITUDE	SUN AZIM ANGLE	LV/LH	
	HR	MIN	SEC	DAY	HR	MIN	SEC	DEG	MIN	DEG	MIN	N. MI.	DEGREES	PITCH DEG	YAW DEG
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
SUNSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
SUNSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
SUNRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX	XXX	XXX	XXX
TERMINATOR RISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
TERMINATOR SET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX			
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Figure 22. Daylight-Darkness Summary Sheet

SPACECRAFT MOON SIGHTINGS

VEHICLE 1

	GET			GMT				GEODETIC LATITUDE		LONGITUDE		ALTITUDE
	HR	MIN	SEC	DAY	HR	MIN	SEC	DEG	MIN	DEG	MIN	N. MI.
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONRISE	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
MOONSET	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XXXX
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Figure 23. Moon Sighting Summary Sheet

COMPUTED EVENTS

VEHICLE 1

	GET HR	GET MIN	GET SEC	GMT DAY	GMT HR	GMT MIN	GMT SEC	ALTITUDE N. MI.	GEODETIC LATITUDE DEG	GEODETIC LATITUDE MIN	LONGITUDE DEG	LONGITUDE MIN	RIGHT ASCENSION HR	RIGHT ASCENSION MIN	INCLINATION DEGREES
ASC NODE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
PERIGEE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX	XX	XX	XXX
REVOLUTION X	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
APOGEE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
ALTITUDE	XXX	XX	XX	XX	XX	XX	XX	XXXX	XX	XX	XXX	XX			
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.	.

Figure 24. Computed Events Summary Sheet

LANDMARK SIGHTINGS

VEHICLE 1

LANDMARK NUMBER	REV NO.	ACQUISITION GET H	ACQUISITION GET M	ACQUISITION GET S	ACQUISITION GMT D	ACQUISITION GMT H	ACQUISITION GMT M	ACQUISITION GMT S	LOSS OF LANDMARK GET H	LOSS OF LANDMARK GET M	LOSS OF LANDMARK GET S	LOSS OF LANDMARK GMT D	LOSS OF LANDMARK GMT H	LOSS OF LANDMARK GMT M	LOSS OF LANDMARK GMT S	DELTA T H	DELTA T M	DELTA T S	ACQ AZ DEG	ACQ GRND RANGE N. MI.	MIN GRND RANGE N. MI.	MAX ELEV DEG	GET OF MAX ELEV H	GET OF MAX ELEV M	GET OF MAX ELEV S	ALT OF MAX ELEV N. MI.
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
XXX	XXX	XXX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	X	XX	XX	XXX	XXXXX	XXXXX	XX	XXX	XX	XX	XXXXX
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Figure 25. Landmark Sighting Summary Sheet

STAR SIGHTINGS

VEHICLE 1

	GET HR	GET MIN	GET SEC	RIGHT ASCENSION HR	RIGHT ASCENSION MIN	RIGHT ASCENSION SEC	DECLINATION DEG	DECLINATION MIN	DECLINATION SEC	MAGNITUDE	CENTRAL ANGLE DEG	CENTRAL ANGLE MIN	ALTITUDE N. MI.	AZIMUTH DEGREES
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX RISE	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
STAR XX SET	XXX	XX	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXX	XX	XXXXX	XXX
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Figure 26. Star Sighting Summary Sheet

CLOSEST APPROACH

VEHICLE 1

TARGET ACR LAND

LATITUDE DEGREES	LONGITUDE DEGREES	ALTITUDE FEET
XXXXX	XXXXX	XXXXX

REV	DAY	GMT HR	GMT MIN	GMT SEC	GET HR	GET MIN	GET SEC	GROUND RANGE N. MI.	ALTITUDE N. MI.	GIMBAL ANGLES PITCH (I) DEG	GIMBAL ANGLES YAW (M) DEG	GIMBAL ANGLES ROLL (0) DEG	LV/LH PITCH DEG	LV/LH YAW DEG
1	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
2	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
3	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
4	XX	XX	XX	XX	XXX	XX	XX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
.	.	.	.	.	.	.	.	.	.	.	.	.	.	.
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Figure 27. Closest Approach Summary Sheet

POINTING DATA

VEHICLE 1

TARGET XXXXX LATITUDE XXXXX DEG LONGITUDE XXXXX DEG ALTITUDE XXXXX N.MI.

REFSMMAT

```
* * * * * * * * * * * * * * * * * * * * * * * * *
*   X   XXXXX   XXXXX   XXXXX   X   *
*   Y   XXXXX   XXXXX   XXXXX   Y   *
*   Z   XXXXX   XXXXX   XXXXX   Z   *
* * * * * * * * * * * * * * * * * * * * * * * * *
```

REVOLUTION XXX

	GET HR : MIN : SEC	GMT DAY : HR : MIN : SEC	ELAPSED TIME MIN : SEC	MAX ELEV ANGLE DEG	MINIMUM RANGE N.MI.	ALT MIN RANGE N.MI.
ACQ	XXX XX XX	XX XX XX XX	XX : XX	XX : XX	XXXXX	XXXXX
LOSS	XXX XX XX	XX XX XX XX				

									LVLH	
GET HR: MIN : SEC	GMT DAY: HR: MIN: SEC	RANGE N.MI.	R-DOT FT/SEC	R-ACC FT/SEC2	AZIMUTH DEG	ELEVATION DEG	ROLL(0) DEG	PITCH(I) DEG	YAW(M) DEG	PITCH DEG	YAW DEG
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXX XX XX	XX XX XX XX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX

Figure 28. Pointing Data Summary Sheet

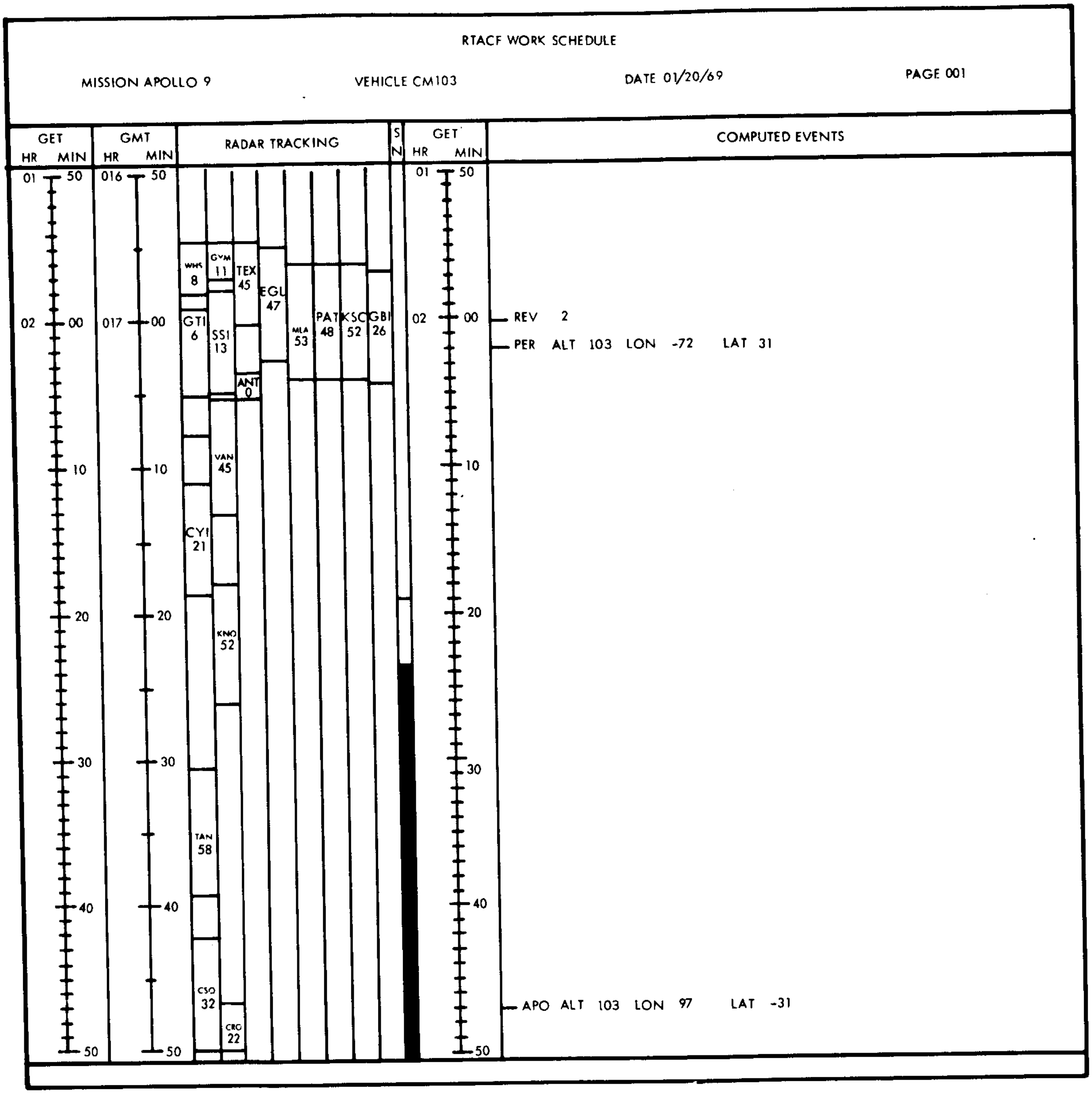

Figure 29. Work Schedule Format

TRIM AERODYNAMIC COEFFICIENTS

XCG = XXXXX YCG = XXXXX ZCG = XXXXX
WEIGHT = XXXXX
BANK ANGLE BIAS = XXXXX DEG.

MACH NO.	ALPHA	CL	CD	CL/CD
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 30. Aerodynamics Update Summary Sheet

MODULES =	1 2 . . . .
X=	XXXXX
Y=	XXXXX
Z=	XXXXX
IX=	XXXXX
IY=	XXXXX
IZ=	XXXXX
WT=	XXXXX

Figure 31. Center of Gravity Summary Sheet

C.G., MOMENTS OF INERTIA, PRODUCT MOMENTS OF INERTIA, AND TRIM ANGLES VERSUS TOTAL WEIGHT

WEIGHT LBS	X INCHES	Y INCHES	Z INCHES	IXX SLUG-FT2	IYY SLUG-FT2	IZZ SLUG-FT2	PXY SLUG-FT2	PXZ SLUG-FT2	PYZ SLUG-FT2	PITCH TRIM DEG.	YAW TRIM DEG.
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXX	XXX	XXX	XXX	XXX
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Figure 32. Mass Properties Summary Sheet

CSM (OR LGC) DAP UPLINK

PRA	ADD	OCTAL	ENG. U.	UPL. U.
IXX	XXXX	XXXXX	XXXXXXX	XXXXXXX
IAVG	XXXX	XXXXX	XXXXXXX	XXXXXXX
WEIGHT	XXXX	XXXXX	XXXXXXX	XXXXXXX
PITCH TRIM	XXXX	XXXXX	XXXXXXX	XXXXXXX
YAW TRIM	XXXX	XXXXX	XXXXXXX	XXXXXXX
TLX	XXXX	XXXXX	XXXXXXX	XXXXXXX

*** XCG = XXX YCG = XXX ZCG = XXX ***

Figure 33. Digital Autopilot Command Load Summary Sheet

SM RCS PROPELLANT COMPUTATION

	QUAD A	QUAD B	QUAD C	QUAD D
TANK	XXX	XXX	XXX	XXX
PH	XXXXX	XXXXX	XXXXX	XXXXX
T	XXXXX	XXXXX	XXXXX	XXXXX
DELT	XXXXX	XXXXX	XXXXX	XXXXX
PO	XXXXX	XXXXX	XXXXX	XXXXX
PF	XXXXX	XXXXX	XXXXX	XXXXX
MR1	XXX	XXX	XXX	XXX
MR2	XXX	XXX	XXX	XXX

	PRI	AUX	PRI	AUX	PRI	AUX	PRI	AUX
WFE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOE	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WFR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WOR	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX
WPU	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX	XXXXX

Figure 34. PVT Summary Sheet

STA ID XXX
GET XXX:XX:XX

INPUT

PLSS PRIMARY OXYGEN SUBSYSTEM DECAY RATE	WO2 =	XXXXXX	LBS/HR
EMU LEAK RATE	O2LR =	XXXXXX	LBS/HR
TOTAL HEAT INTO THE SYSTEM FROM EQUIPMENT OPERATION	QEQ =	XXX	BTU/HR
TOTAL HEAT INTO THE SYSTEM FROM ENVIRONMENT	QLEAK =	XXX	BTU/HR
HEAT LEAK FROM SUIT ENVIRONMENT	QSS =	XXX	BTU/HR
HEAT FROM LUNGS, PRESPIRATION, RESPIRATION, ETC	QR =	XXX	BTU/HR
SUBLIMATOR LOOP MASS FLOW RATE	M1 =	XXX	LBS/HR
LCG H_2O INLET TEMP	T =	XXX	DEG F
LCG H_2O OUTLET TEMP	T2 =	XXX	DEG F
TOTAL H_2O TRANSPORT LOOP MASS FLOW RATE	M2 =	XXX	LBS/HR
EFFICIENCY NUMBER	E =	XXX	
HEAT OUTPUT FROM THE REACTION OF CO2 WITH LIOH	CF =	XXXXXX	BTU/LB
FEED WATER PRESSURE	FW PRESS =	X. XX	

OUTPUT

	THR	FWUR
MR	XXXX	X.XX
LE	XXXX	X.XX
FP	XXXX	X.XX

Figure 35. EMU Summary Sheet

LM - RCS PROPELLANT BUDGET

TIME HRS	M	EVENT TITLE	S/C WT (LBS)	LM RCS USED (LBS)	LM RCS LEFT (LBS)	LM RCS LEFT PCNT
XXX	XX	OUTPUT PROPELLANT LOADINGS	XXXXX	XX	XXX	XXX
XXX	XX	REDLINE OFF	XXXXX	XX	XXX	XXX
XXX	XX	RCS HOT FIRE	XXXXX	XX	XXX	XXX
XXX	XX	PGNS 3AX ATTITUDE HOLD 1DB	XXXXX	XX	XXX	XXX
XXX	XX	PITCH	XXXXX	XX	XXX	XXX
XXX	XX	YAW	XXXXX	XX	XXX	XXX
XXX	XX	PGNS 3AX ATTITUDE HOLD 1DB	XXXXX	XX	XXX	XXX
XXX	XX	PITCH	XXXXX	XX	XXX	XXX
XXX	XX	YAW	XXXXX	XX	XXX	XXX
XXX	XX	2 JET ULLAGE	XXXXX	XX	XXX	XXX
XXX	XX	YAW MOMENT CONTROL DOCKED BURN	XXXXX	XX	XXX	XXX
XXX	XX	DOCKED DPS BURN	XXXXX	XX	XXX	XXX
•	•	•	•	•	•	•
•	•	•	•	•	•	•

Figure 36. LM RCS Summary Sheet

SUPERCRITICAL HELIUM PROFILE

GMT D	GMT H	GMT M	GMT SEC	TIME GET H	TIME GET M	TIME GET SEC	GET SEC	THRUST (LBS)	BOTTLE PRESSURE (PSI)	BOTTLE MASS (LBS)	BOTTLE TEMP (DEG F)	REG OUT TEMP (DEG F)	PROP WT (LBS)
XX	XX	XX	XX	XXX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XXX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XXX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
XX	XX	XX	XX	XXX	XX	XX	XXXXXX	XXXXX	XXXX	XXXX	XXXX	XXX	XXXXXX
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THE MAXIMUM PRESSURE OF XXXX.X PSI, OCCURS AT GET XXXXXX.X SECONDS.

Figure 37. SHe Summary Sheet

LM TELEMETRY DIAGNOSTICS

THERE ARE 2 FAILURES ASSOCIATED WITH SEQUENCER 7 PAM GATE NO 1, THEY ARE
GC0202 GC0204
SEQUENCER 7 PAM GATE NO 1 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS
GC0202 GC0204
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 7 PAM GATE NO 2, THEY ARE
GF3585
SEQUENCER 7 PAM GATE NO 2 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS
GF3585
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 9 PAM GATE NO 1, THEY ARE
GC0203
SEQUENCER 9 PAM GATE NO 1 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS
GC0203 GF3571
THERE ARE 1 FAILURES ASSOCIATED WITH SEQUENCER 14 PAM GATE NO 3 , THEY ARE
GC0155
SEQUENCER 14 PAM GATE NO 3 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS
GC0155 GR1085
THERE ARE 3 FAILURES ASSOCIATED WITH ROW 3. THEY ARE
GF3585 GC0202 GC0204
ROW 3 IS ASSOCIATED WITH THE FOLLOWING PARAMETERS
GC0202 GC0204 GF3585 GQ3015 GQ3018

Figure 38. LM TM Diagnostics Summary Sheet

VECTOR IDENTIFICATION

EVENT TIMES	RET			GMT				GET			LATITUDE		LONGITUDE	
	HR	MIN	SEC	DAY	HR	MIN	SEC	HR	MIN	SEC	DEG	MIN	DEG	MIN
RETROFIRE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BURN TERMINATION	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
ENTRY INTERFACE	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
.20 G'S	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
REVERSE BANK	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, VHF	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
END BLACKOUT, S BAND	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
DROGUE DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
MAIN DEPLOY	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX
LANDING	XX	XX	XX	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XXX	XX

BURN QUANTITIES

DELTA VELOCITY(X)	XXXXX	FT/SEC
DELTA VELOCITY(C)	XXXXX	FT/SEC
DELTA VELOCITY(T)	XXXXX	FT/SEC
DELTA TIME	XXXXX	SEC
WEIGHT	XXXXX	POUNDS
TRUE ANOMALY	XXXXX	DEG
THRUST PITCH	XXXXX	DEG
LATITUDE	XX DEG	XX MIN
LONGITUDE	XXX DEG	XX MIN
ALTITUDE	XXXXX	N MI
EXTERNAL DELTA V		
DX	XXXXX	FT/SEC
DY	XXXXX	FT/SEC
DZ	XXXXX	FT/SEC

GIMBAL ANGLES	OUTER (ROLL)	INNER (PITCH)	MIDDLE (YAW)
BURN	XXX	XXX	XXX
ENTRY INTERFACE	XXX	XXX	XXX
REVERSE BANK, BEGIN	XXX	XXX	XXX
REVERSE BANK, END	XXX	XXX	XXX

REENTRY QUANTITIES

VELOCITY(I) EI	XXXXX	FT/SEC
GAMMA(I) EI	XXXXX	DEGREES
WEIGHT	XXXXX	POUNDS
BANK ANGLE	XXXXX	DEG
LIFT	XXXXX	
MAX G'S	XXXXX	

LANDING POINTS

LATITUDE	TARGET	XX DEG	XX MIN
LONGITUDE	TARGET	XXX DEG	XX MIN
LATITUDE	IMPACT	XX DEG	XX MIN
LONGITUDE	IMPACT	XXX DEG	XX MIN

Figure 39. Standard Summary Sheet

SUMMARY SHEET - PRE-BURN ENTRY

EVENT TIMES	GMT				GET			RET			LATITUDE		LONGITUDE	
	DAY	HR.	MIN.	SEC.	HR.	MIN.	SEC.	HR.	MIN.	SEC.	DEG.	MIN.	DEG.	MIN.
RETROFIRE	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BURN TERMINATION	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
SEPARATION	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
ENTRY INTERFACE	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT (VHF)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
BEGIN BLACKOUT (S)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
.05G'S(EMS)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
0.2G'S	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
REVERSE BANK	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
END BLACKOUT (S)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
END BLACKOUT (VHF)	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
DROGUE DEPLOY	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
MAIN DEPLOY	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX
LANDING	XX	XX	XX	XX	XXX	XX	XX	XX	XX	XX	XX	XX	XXX	XX

BURN QUANTITIES

DELTA VELOCITY	=	XXXXX	FT/SEC		
DELTA TIME	=	XX	MIN	XX	SEC
WEIGHT	=	XXXXX	POUNDS		
PERIGEE RESULTING	=	XXXXX	N. MILES		
TRUE ANOMALY	=	XXXXX	DEGREES		
ALTITUDE	=	XXXXX	N. MILES		
LATITUDE	=	XX	DEG	XX	MIN
LONGITUDE	=	XXX	DEG	XX	MIN
EXTERNAL DELTA VX	=	XXXXX			
VY	=	XXXXX			
VZ	=	XXXXX			

GIMBAL ANGLES	OUTER (ROLL)	INNER (PITCH)	MIDDLE (YAW)
BURN	XXX	XXX	XXX
SEPARATION	XXX	XXX	XXX
400K	XXX	XXX	XXX
300K	XXX	XXX	XXX
BACKUP BANK AT .20G	XXX	XXX	XXX
BACKUP BANK AT RB	XXX	XXX	XXX

LANDING POINTS

	LATITUDE		LONGITUDE	
	DEG.	MIN.	DEG.	MIN.
MIN LIFT	XX	XX	XXX	XX
MAX LIFT	XX	XX	XXX	XX
TARGET	XX	XX	XXX	XX
G AND N	XX	XX	XXX	XX
BACK UP	XX	XX	XXX	XX

REENTRY QUANTITIES

INITIAL BANK			XXX	DEG(RIGHT)
BACKUP BANK			XXX	DEG(LEFT)
RETRB	XX	MIN	XX	SEC
RET400K	XX	MIN	XX	SEC
RET.05G	XX	MIN	XX	SEC
RET.2G	XX	MIN	XX	SEC
RET.XXG	XX	MIN	XX	SEC
RETBBO (VHF)	XX	MIN	XX	SEC
RETEBO (VHF)	XX	MIN	XX	SEC
RETDD	XX	MIN	XX	SEC
RETMD	XX	MIN	XX	SEC
RT400K			XXXXX	N.MILES
V400K			XXXXX	FT/SEC
GAMMA400K			XX	DEG
(RP-RT).2G			XXX	N.MILES
V.05G			XXXXX	FT/SEC
RNG.05G			XXXXX	N.MILES
RETEMS	XX	MIN	XX	SEC
VEMS			XXXXX	FT/SEC
RNGEMS			XX	N.MILES
MAXG			XX	
L/D			XXXX	
WEIGHT			XXXXX	POUNDS

Figure 40. ARS Summary Sheet

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